NPS ARCHIVE
1961.06
SIMS, G.

AN INVESTIGATION OF THE INFLUENCE OF ALTITUDE RESPONSE ON THE PILOT'S SELECTION OF APPROACH SPEED

G. L. SIMS
and
D. G. W. TERRY

LIBRARY
U.S. NAVAL POSTGRADUATE SCHOOL
MONTEREY, CALIFORNIA

AN INVESTIGATION OF THE INFLUENCE OF ALTITUDE RESPONSE ON THE PILOT'S SELECTION OF APPROACH SPEED

G. L. Sims, Jr., Lieutenant, USN
D. G. W. Terry, Lieutenant, USN

Aeronautical Engineering Department

Report No. 552
May 1961

Submitted in partial fulfillment of the requirements
for the Degree of Master of Science in Engineering
from Princeton University, June 1961

Library
U. S. Naval Postgraduate School
Monterey, California

FOREWORD

This report represents the culmination of an interest aroused in the authors following a discussion with Professors Edward Seckel and Dunstan Graham at Princeton University in October 1960. The problem had particular appeal to the authors, especially those aspects which might be related to landing approaches made on the Mirror Landing System in use by the Navy.

The authors are indebted to the many persons whose aid and encouragement contributed greatly to the successful completion of the project. In particular, they wish to express their appreciation to Professor Edward Seckel, under whose direction the project was undertaken; Professor D. H. Graham, for his comments on theoretical problems; Mr. E. J. Durbin, whose many helpful suggestions and aid in equipment installation were invaluable; Mr. Richard Whitley and Mr. William Szabelski for their work in installation and maintenance of the necessary equipment; and to the many others whose contributions in some measure aided in bringing the project to completion. The authors particularly wish to extend their thanks to Mrs. Grace Arnesen, whose excellent job of manuscript typing is evident herein.

TABLE OF CONTENTS

Page

FOREWORD i

TABLE OF CONTENTS ii

LIST OF SYMBOLS iv

SUMMARY vi

INTRODUCTION vii

BACKGROUND 1

ANALYSIS 3

- Discussion of Transfer Functions 5
- Steady State Response 8
- Variation of Altitude Response with Throttle Feedback 10
- Velocity Feedback 12
- Angle of Attack Feedback 13
- Pitch and Pitch Rate Feedbacks 14
- $h/-\delta_e$ with Varying $C_{D_{eff}}$ 16
- Transient Response 19
- Computer Study 21
- Altitude Control and the Thrust Required Curve 23

EQUIPMENT 30

CALIBRATION 32

PROCEDURE 33

DISCUSSION OF RESULTS 35

CONCLUSIONS 41

RECOMMENDATIONS 43

REFERENCES 44

Page

APPENDIX

A. Stability Derivatives 45
B. Development of Transfer Functions 47
C. $h/-\delta_e$ with Varying $C_{D_{eff}}$ 52
D. Computer Mechanization 54
E. Development Throttle Control Effectiveness 55

TABLES AND FIGURES

LIST OF SYMBOLS

A	Aspect ratio
$\bar{c}$	Mean aerodynamic chord, feet
C_{D_α}	Partial derivative of drag coefficient with respect to angle of attack
$C_{D_{eff}}$	Effective drag coefficient
C_{D_o}	Steady state induced and profile drag coefficient
C_{L_α}	Partial derivative of lift coefficient with respect to angle of attack
C_{L_o}	Steady state lift coefficient
C_{m_α}	Partial derivative of moment coefficient with respect to angle of attack
$C_{m_{d\alpha}}$	Partial derivative of moment coefficient with respect to rate of change of angle of attack
$C_{m_{d\theta}}$	Partial derivative of moment coefficient with respect to pitch rate
$C_{m_{\delta_e}}$	Partial derivative of moment coefficient with respect to elevator deflection
C_{m_u}	Partial derivative of moment coefficient with respect to nondimensional velocity change
$C_{T_{\delta_e}}$	Partial derivative of thrust coefficient with respect to elevator deflection
$C_{T_{\delta_t}}$	Partial derivative of thrust coefficient with respect to throttle movement (Throttle effectiveness parameter)
d( )	Operator d( )/d(t/r)
e	Oswald airplane efficiency factor
h	Variation in altitude, feet
$\bar{h}$	Airplane inertia factor
k_y	Radius of gyration, feet

m	Airplane mass, slugs
N	Transfer function numerator
S	Wing area, feet^2
t	Time, seconds
u	Nondimensional perturbation velocity
V	Velocity, feet/second
α	Angle of attack, radians
$\boldsymbol{\Delta}(s)$	Denominator of a transfer function
$\Delta(s)$	Denominator of a transfer function in normalized form
θ	Pitch angle, radians
μ	Airplane nondimensional density factor
ρ	Density, slugs/ft.^3
τ	Nondimensional time, airseconds
ζ	Damping ratio
ω	Frequency, radians/second
ω_n	Undamped natural frequency, radians/second
δ_e	Elevator displacement, radians
δ_t	Throttle displacement, inches
(s)	Laplace transform variable

AN INVESTIGATION OF THE INFLUENCE
OF ALTITUDE RESPONSE ON THE PILOT'S SELECTION
OF APPROACH SPEED

SUMMARY

An investigation of the factors which influence the pilot's choice of landing approach speed was conducted by means of a theoretical study, computer program and flight evaluation. In particular, the effect of the altitude response of the airplane at low airspeeds was investigated. Results obtained indicated that the altitude response and phugoid damping played a large part in determining the pilot's opinion of the flying qualities of the airplane in slow flight. While the study did cover what were considered the salient points in the theoretical and computer portions of the investigation, time and equipment limitations precluded a thorough flight study of all characteristics.

The study was conducted at Forrestal Research Center, Princeton University, Princeton, New Jersey, beginning in November 1960 and ending in May, 1961.

INTRODUCTION

The most critical phase of airplane flight, from the aspect of safety, is the landing, and the basis of a good landing is a properly executed approach. In the approach, precise control of the airplane in the longitudinal modes of motion is necessary. This precise control allows the pilot to fly a predetermined track over the ground, and systematically decrease altitude to the point of touchdown. During the latter stages of the approach, altitude control is most critical.

In this report, an investigation of the primary factors which affect the altitude control of the airplane have been studied. These factors have then been analyzed, and an attempt made to show how they may be utilized, or modified, to allow more precise control of the airplane during the landing approach, and the subsequent effect on the pilot's opinion of handling qualities.

The analysis was performed by first making a theoretical study of the problem and compiling the important factors. Next, a computer study was made to obtain verification of the theoretical results, and finally, a flight test of some of the important factors was carried out in an instrumented airplane.

The study was performed at the Forrestal Research Center of Princeton University during the period November 1960 to May 1961.

BACKGROUND

The trend of modern airplanes toward higher speeds and increased weight has intensified the need for longer runways or a means of safely landing at a slower speed. The landing rollout necessary can be reduced considerably by utilizing a slower approach speed. The slowest speed which can be used in the landing approach is often dependent on the airspeed for which minimum thrust is required. Below this airspeed, the airplane is on the "backside of the curve", where an increase in velocity results in a decrease in drag.

Before the advent of the Mirror Landing System, Naval Aviators flew approaches on the "backside" of the thrust required curve. This technique required a considerable amount of practice before the pilot became proficient. With the Mirror System now in use, speeds above the minimum thrust speed may be used. Studies of pilots flying such a precision approach reveal that they tend to fly above the recommended speed. This analytic study concerns altitude and airspeed control at airspeeds in the vicinity of that for minimum thrust required. This data has been further analyzed in an effort to determine why the pilot exhibits a tendency to fly the landing approach at higher than recommended airspeeds.

In most precision landing approaches, glide slope information is presented to the pilot as an altitude error. This is true of the Navy Mirror Landing System, Instrument Landing System and Ground Controlled Approach System. In

order to understand the relationship between elevator and throttle control, and the resultant trade-off between altitude and airspeed, the relation between the kinetic and potential energies of the airplane must be examined.

In the landing approach, the pilot is concerned primarily with control of the airplane altitude and airspeed in order to follow a desired track over the ground, and systematically decrease the altitude to the point of touchdown. In performing this task, he has two primary controls available (neglecting such devices as trim panels, speed brakes, etc.). These controls are the elevator and throttle. The airplane maintains flight due to possession of a certain energy. This energy is introduced into the system through the engine, and the rate of introduction is controlled by the throttle. When the rate of introduction of energy equals the rate of dissipation of energy, the total energy level is held constant, and an equilibrium flight condition is maintained. When the rate of introduction of energy into the system exceeds the rate of dissipation, the airplane will either accelerate in level flight, or climb, or will do both. The opposite is true when the rate of dissipation of energy exceeds the rate of addition. The elevator, as opposed to the throttle, is merely a means for conversion of energy from one form to another, and cannot change the total energy level. Assume for the moment the case of equilibrium flight, and it may easily be seen that the movement of the elevator, through its effect on the attitude, will result in a conversion of potential to kinetic or kinetic to potential energy.

It will also be noted that speed changes affect the energy level of the airplane. For operation at a given airspeed, in level flight, a certain drag is experienced. Disturbance from this speed will either increase or decrease the drag which the airplane experiences. If the drag is decreased, the airplane will tend to climb at the new airspeed or to return to the original airspeed. If the drag is increased, the airplane will tend to descend at the new airspeed or will return to the original airspeed.

During the normal landing approach, the pilot flies very near the minimum airspeed (kinetic energy level) consistent with safe operation. He maintains this airspeed through proper manipulation of the elevator. To increase or decrease the altitude (potential energy level), the pilot uses the throttle. This method of control may be summed as follows: Throttle controls altitude and elevator controls airspeed.

ANALYSIS

The first step in analyzing the control problem at approach airspeeds was the solution of the longitudinal equations of motion and the development of the longitudinal transfer functions. Two control inputs were considered, the elevator and the throttle.

The longitudinal equations as used are

$$\begin{aligned}
(C_D + d)u &+ \tfrac{1}{2}(C_{D_\alpha} - C_L)\alpha + \tfrac{1}{2}C_L\theta = C_{T_{\delta_t}} \\
C_L u &+ (C_{L_\alpha}/2 + d)\alpha - d\theta = 0 \\
0 &+ (C_{m_\alpha} + C_{m_{d\alpha}}d)\alpha + (C_{m_{d\theta}}d - \bar{h}d^2)\theta = -C_{m_{\delta_e}}\delta_e
\end{aligned}$$

The stability derivatives for the Navion at 80 mph and 4500 ft. density altitude are derived in Appendix A. Using these derivatives the longitudinal equations of motion become

$$(.092+d)u + (-.1675)\alpha + .52\theta = .0895\delta_t$$

$$1.04u \quad + (2.725+d)\alpha - d\theta = 0$$

$$0 \qquad + (-.485 - .0745d)\alpha + (-.1595d - .0532d^2)\theta = -1.435\delta_e$$

The characteristic equation as solved in Appendix B is of the form

$$C_4(\lambda^4 + C_3/C_4\,\lambda^3 + C_2/C_4\,\lambda^2 + C_1/C_4\,\lambda + C_0/C_4) = 0$$

where

$$C_4 = -.0532$$
$$C_3 = 7.20$$
$$C_2 = 18.1$$
$$C_1 = 2.87$$
$$C_0 = 4.93$$

The solution of the altitude was desired as altitude must be accurately controlled during approaches. Also the pilot observes his glide slope error as an altitude error.

At small glide slope angles

$$\dot{h} = V\sin\gamma \cong V\gamma$$

since $\gamma = \theta - \alpha \quad ; \quad \dot{h} = V(\theta - \alpha)$

then $h = \int \dot{h}dt = V(\theta - \alpha)dt.$

Using Laplace transforms

$$h(s) = V(\theta - \alpha)/s$$

The solutions of the altitude transfer functions are contained in Appendix B, and the results tabulated with the other transfer functions in Table I.

The elevator transfer functions are based on negative elevator motion, as up elevator is required to increase all the variables except velocity.

The Bode diagrams of these transfer functions are Figures 1 through 10.

Three methods of examining the transfer functions were used. These were examinations of:

1. The Bode diagrams and the root loci of the transfer function with unity feedback.
2. The steady state responses due to step control inputs.
3. Transient response of the open loop transfer function.

The purpose of these examinations was to seek some key as to why pilots apparently prefer faster approach speeds.

DISCUSSION OF THE TRANSFER FUNCTIONS BASED ON BODE DIAGRAMS AND ROOT LOCI

This discussion is based on the Bode diagrams and root loci plots, Figures 1 through 20, for the longitudinal transfer functions. The pilot is assumed to act as a pure gain unless otherwise noted.

Table II tabulates the data which is available in the root loci. Taking the transfer functions in the order listed.

U/δ_e: Positive elevator is used in this transfer function only, as positive elevator gives an increase in airspeed. The pilot must use a lead-lag to prevent phugoid instability at a low gain. The short period damping is increased by closing the loop. The Bode shows the phugoid motions can be controlled at a low gain. The finite d.c. gain shown in the Bode shows the trim speed change due to change in elevator position.

$\alpha/-\delta_e$: This transfer function is stable for all gain. The pilot can act as pure gain and quite easily control short period motion. In this case he can even control the phugoid. The phugoid could not be so well controlled at lower lift coefficients with this α feedback. Closing the loop has little effect on the phugoid but decreases the short period damping. The finite d.c. gain again shows the trim change cuased by a change in the elevator position.

$\theta/-\delta_e$: This transfer function is also stable for all gain. The phugoid damping is increased but the short period damping, decreased. The Bode shows that pitch angle feedback can be used to control the phugoid and the short period, however the d.c. gain is low and diminishes as one of the numerator zeros approaches zero. This low d.c. gain, compared to the short period gain may have some influence on aircraft handling qualities at low speed.

$\dot{h}/-\delta_e$: This transfer function is stable for all but high gains. Closing the loop increases the phugoid damping. Rate of climb feedback should be able to

control the phugoid. Instrument lags, which are not shown in the transfer function, would be detrimental at higher frequencies. The finite d.c. gain again shows the change of trim associated with changes in the elevator position.

$h/-\delta_e$: This transfer function shows there is a pure divergence for any gain. As discussed later, however, this pure divergence can be eliminated in the pilot switches wires, that is if he uses negative gain. If he should do this, the phugoid would become unstable at a low gain. The pilot then would have to use a lead-lag so that he could increase the gain. This transfer function will be discussed more fully later.

U/δ_t: The root locus shows this transfer function is stable for all gain. The pilot can be pure gain and should be quite happy with this feedback. Note that the d.c. gain is zero. This implies that steady state speed changes can not be made with the throttle.

α/δ_t: The phugoid becomes unstable at low gain, showing the destabilizing effect of power on the phugoid. The short period damping is however increased. The zero d.c. gain again shows the impossibility of effecting steady state speed changes with the throttle.

θ/δ_t: As with the previous transfer function, this one shows an unstable phugoid at low gain. The pilot could control the phugoid if he were to provide a lead-lag. The finite d.c. gain shows that changing the throttle setting changes the flight path angle since $\alpha_{d.c.}$ is unchanged by the throttle.

$\dot{h}/\delta_t$: This transfer function is stable for all gain and should be very favorable for the pilot to use. The short period roots are changed very little on closing the loop. The phugoid period is decreased but the damping relatively unaltered. The finite d.c. gain shows that the steady state rate of climb is proportional to the throttle setting.

h/δ_t: This transfer function shows that for all but very small gain the phugoid is unstable. The pilot would have to use negative gain to stabilize the phugoid. The high d.c. gain is the result of integrating the rate of climb.

The pilot can use the transfer functions discussed above either singly or in combination. Normally he must use more than one to correct glide slope or airspeed errors. Step control inputs may be assumed to be used to correct for glide slope and airspeed errors.

An airspeed error would normally be corrected with elevator deflection, while the glide slope is maintained with the throttle. On the other hand, a glide slope error would be corrected with the throttle while airspeed is maintained with the elevator. Should there be both a glide slope and an airspeed error, the correction could be a trade off, as if high and slow or low and fast, or the correction could be made to first one and then the other.

EXAMINATION OF THE STEADY STATE RESPONSE TO STEP INPUTS

The steady state responses to step inputs for the various transfer functions are shown in Table III. The results show that in general, an increase in

airspeed reduces the magnitude of the responses, but does not alter the character of the response. The exceptions to this are the altitude and rate of climb responses. These two quantities show a reversal of sign dependent upon whether the airplane is operating above or below minimum throttle setting. In addition, as altitude is the integral of rate of climb, this goes either to plus or minus infinity. As the altitude to elevator response shows such a marked change due to velocity changes, this investigation was guided toward an examination of this transfer function. The study of $h/-\delta_e$ was not concerned with a desire to control altitude with elevator, and in fact, Table III shows that it is impossible to control altitude in this manner. This is due to the necessity of using the elevator to control airspeed, which with reference to the trade-off of kinetic and potential energies previously discussed, necessitates the use of throttle for altitude control. This is a treatment of the matter which ignores the transient responses (discussed in a later section), the effect of power on lift and moment, and the effect of the elevator on lift. This omission is partially justified on the basis of limiting the investigation to small changes of elevator and power, over which range their effect may be ignored in the approximation.

Examination of the transfer function $h/-\delta_e$ shows that the major difference in its nature due to velocity changes in the vicinity of approach speeds, can be related to the numerator term. The numerator denotes a nonminimum phase condition for airspeeds below that for minimum thrust required, and minimum phase for airspeeds greater than that for minimum thrust required. This movement

of airspeed from the backside to the frontside of the thrust required curve is evidenced in the steady state response by the reversal of sign previously noted. A throttle servo was used to change the numerator term without changing the airspeed, in order to investigate the effect of movement of the zero of the transfer function at a constant airspeed. By changing the transfer function in the same manner as would be done by an increasing airspeed, yet holding velocity constant, it was hoped to gain knowledge of the effect of this transfer function on the choice of approach speed by the pilot. In this study, other possible contributory effects such as the lateral response and phsycological effects were not varied.

In the following sections, several methods of changing the numerator term are discussed. All of these methods use a feedback through a throttle servo, this being the only means available to modify the drag equation.

VARIATION OF THE AIRPLANE ALTITUDE RESPONSE WITH THROTTLE FEEDBACK

The character of the altitude response of the airplane to an elevator deflection is largely determined by the position of the transfer function zero. For the transfer function $h/-\delta_e$, this zero is $\left(S + C_D + \frac{C_{D_\alpha} C_L}{C_{L_\alpha}} \right)$. This zero position corresponds to a position on the thrust required curve for reaction type engine configurations and to a position on the power required curve for reciprocating engine configurations. The numerator term will vanish when the sum of the three components equal zero. This corresponds to the minimum throttle position.

The position of the zero under discussion is $\frac{C_L C_{D\alpha}}{C_{L\alpha}} - C_D$. This zero is nonminimum phase when $C_D < \frac{C_L C_{D\alpha}}{C_{L\alpha}}$. When the zero becomes minimum phase, the direction of the response is changed. Thus for $C_D < \frac{C_L C_{D\alpha}}{C_{L\alpha}}$, the steady state response is $>$ 0, and for $C_D > \frac{C_L C_{D\alpha}}{C_{L\alpha}}$, the steady state response is $<$ 0. The significance of the zero position is readily apparent by reducing $\frac{C_L C_{D\alpha}}{C_{L\alpha}}$ to $2\, C_{D_i}$, and setting $C_D = C_{D_f} + C_{D_i}$. The zero position is therefore dependent on the thrust required curve, and is minimum phase when the airplane is operating on the front side of the thrust required curve. A similar development may be made for the reciprocating engine configuration to show that the zero changes from nonminimum to minimum phase at the minimum power required point. Both cases however can be referred to as minimum throttle points. In the succeeding development, the development will be limited to the reaction engine configuration, however, a similar approach may be applied to the reciprocating engine configuration.

Obviously, a nonminimum phase zero may be made minimum phase simply by flying at a greater airspeed. If however, an autopilot is used to change the zero position, then the aircraft may be flown on the backside of the thrust required curve while its response will correspond to operation on the frontside of the curve. This condition may be highly desirable for approaches to short fields, and other precision approaches. Landing at lower airspeeds impose smaller loads on the structure, as well as allowing shorter landing rollout. Various methods were examined for changing the zero position, all utilizing an autopilot to drive

a throttle servo. The feedback quantities examined include velocity, angle of attack, elevator position, pitch angle and pitch rate. The discussion of these follow, and the results are shown in Appendix B.

VELOCITY FEEDBACK

If the airplane thrust varies with velocity, the Drag Equation may be written:

$$\left(C_D - \frac{\partial T/\partial V}{\rho S V} + s\right)u + \frac{1}{2}\left(C_{D_\alpha} - C_L\right)\alpha + \frac{1}{2} C_L \theta = 0$$

If $C_{D_{eff}}$ is defined as $\left(C_D - \frac{\partial T/\partial V}{\rho S V} \right)$, then in the airplane transfer function, $C_{D_{eff}}$ may be substituted for C_D, and the effect of the velocity feedback noted.

The zero of the transfer function will be minimum phase if $C_{D_{eff}} > \frac{C_L C_{D_\alpha}}{C_{L_\alpha}}$. $C_{D_{eff}}$ can be increased by reducing the thrust with increasing velocity (making $\partial T/\partial V$ negative). The root locus plot of the zero for varying $C_{D_{eff}}$ is shown in Figure 21.

The characteristic equation is also altered by variation of $C_{D_{eff}}$, as shown in Appendix C. The locus of the roots of the characteristic equation for varying $C_{D_{eff}}$ is shown in Figure 22. From this plot, it is evident that with increasing $C_{D_{eff}}$, the phugoid becomes more heavily damped. It is also evident that the short period mode is relatively unchanged. These results correspond to the results of the basic analysis of Reference 1. This shows that the short period takes place at essentially constant velocity, whereas relatively large velocity changes are associated with the phugoid. The increase in phugoid

damping exhibited with increasing $C_{D_{\text{eff}}}$ means that the airplane will have less velocity and altitude variation for the case of increased damping, and the pilot may control the flight path with less control manipulation.

ANGLE OF ATTACK FEEDBACK

If angle of attack is fed back to the throttle, the Drag Equation becomes:

$$(C_D + d)u + \left[\tfrac{1}{2}(C_{D_\alpha} - C_L) - C_{T_\alpha}\right]\alpha + \tfrac{1}{2} C_L \theta = 0$$

If $C_{D_{\alpha\,\text{eff}}}$ is defined as $C_{D_\alpha} - 2\,C_{T_\alpha}$, then the same type of analysis as was used with the velocity feedback may be utilized. The zero of the transfer function $h/-\delta_e$ then becomes minimum phase when $C_{D_{\alpha\,\text{eff}}} < \frac{C_D C_{L_\alpha}}{C_L}$. $C_{D_{\alpha\,\text{eff}}}$ is made small by increasing the thrust with increasing angle of attack ($C_{T_\alpha} > 0$). The root locus of the zero is shown in Figure 21, and the root locus of the characteristic equation is shown in Figure 23, for varying $C_{D_{\alpha\,\text{eff}}}$. It should be noted that unlike the velocity feedback, the angle of attack feedback causes an unstable phugoid. The degree of divergence is not so great as to be uncontrollable by the pilot however. The short period also changes considerably with angle of attack feedback.

ELEVATOR POSITION FEEDBACK

With elevator position feedback to the throttle, the Drag Equation becomes:

$$(C_D + d)u + \tfrac{1}{2}(C_{D_\alpha} - C_L)\alpha + \tfrac{1}{2} C_L \theta = C_{T_{\delta_e}} \delta_e$$

The altitude to elevator transfer function for this condition may be written as

$$\left(h/-\delta_e\right)^* = \left(h/-\delta_e\right)_{C_{T_{\delta_e}}=0} + \Delta h/-\delta_e$$

where $\left(h/-\delta_e\right)_{C_{T_{\delta_e}}=0}$ is the transfer function with no elevator feedback, and

$\Delta h/\delta_e$ is the addition due to elevator feedback. Expanding the numerator of the above transfer function (including elevator feedback),

$$N = -\frac{V}{2} C_{m_{\delta_e}} C_{L_\alpha}\left[s + C_D - C_L \frac{C_{D_\alpha}}{C_{L_\alpha}}\right] - V C_{T_{\delta_e}} C_L \bar{h}\left[s^2 + \left(\frac{C_{m_{d\alpha}} - C_{m_{d\theta}}}{\bar{h}}\right)s + \frac{C_{m_\alpha}}{\bar{h}}\right]$$

The root locus for this numerator is shown in Figure 21. Although a nonminimum zero still exists, the sign of the transfer function is changed so that when one zero becomes minimum phase, the steady state response to a step input is < 0, and the airplane responds as if it were on the front side of the thrust required curve.

Physically this result is obvious as the feedback of elevator position provides a means of increasing the thrust as the angle of attack is increased. While the increased angle of attack does increase the lift, it also increases the induced drag, and the feedback provides a means of effectively cancelling this drag increase with thrust. The result is response of the airplane to elevator deflection as if it were on the front side of the thrust required curve.

The characteristic equation of the airplane is unaltered by the feedback, as the change of thrust is due only to the control deflection

PITCH ANGLE AND PITCH RATE FEEDBACKS

The effect of pitch feedback to the throttle may be found by the addition of a term $-C_{T_\theta}$ to the Drag Equation. If the numerator of the transfer function $h/-\delta_e$ is expanded, this gives the addition of a constant term to the original transfer function. The root locus of this modified numerator is similar to that for velocity feedback. However, the roots of the characteristic equation are changed much more than with velocity feedback.

If pitch rate is fed back to the throttle, a term $-C_{T_{\delta\theta}}\delta$ must be added to the Drag Equation. The root locus of the expanded numerator of the $h/-\delta_e$ transfer function shows that a nonminimum phase condition is always associated with the zero.

Following a thorough examination of the foregoing methods of modifying the airplane response, it was decided to utilize the velocity feedback. The velocity feedback, it will be recalled, changes the numerator in the same manner as an increase in approach velocity, while increasing the phugoid damping. In addition, the short period mode was not changed significantly.

A favorable pilot opinion was sought for the airplane response to elevator deflection at various locations of the numerator zero. The purpose of such opinion was to establish some correlation between the pilot's exhibited preference for a higher approach speed on the basis of

1. More favorable altitude response at a higher airspeed
2. A more heavily damped phugoid.

No other method was utilized in the flight test program for changing the response, however, because of the possibility of altering the initial response, the effect of elevator position feedback was investigated in the computer analysis (see discussion of computer analysis).

ANALYSIS OF $h/-\delta_e$ WITH VARYING $C_{D_{eff}}$

For analytic work, five values of $C_{D_{eff}}$ were chosen. These were:

$C_{D_{eff}} = 0$

$= .092$ (basic airplane)

$= .1345$ (minimum thrust required)

$= .24$

$= .334$

The first value gives an altitude to elevator response comparable to that exhibited by an airplane operating on the back side of the thrust required curve, and having an unstable phugoid mode. The second value is that exhibited by the basic Navion. The third value corresponds to operation at the velocity for minimum thrust required. The last two values correspond to operation on the front side of the thrust required curve. The basic Navion and the succeeding conditions all exhibit stable phugoid modes.

The Bode diagrams of the transfer function for different values of $C_{D_{eff}}$ were examined with the intent of gaining information to enable the analyst to predict what the pilot's opinion of the flying qualities of the airplane would be under the different conditions. In particular, it was desired to determine if the feedback of airspeed to the servo throttle would be beneficial to the pilot in exercising precise altitude control. The Bode diagrams are shown in Figure 1.

The first obvious fact on the Bode plots is that at high frequencies, changing the effective drag coefficient has no effect on the response. This information is also shown on the root locus plot by the very close coincidence of the poles and zeros in the left hand half plane. The Bode plots also show that for drag coefficients less than .1345, the steady state response is in the opposite direction to the steady state response for drag coefficients greater than .1345. The coefficient .1345 corresponds to the airspeed for minimum thrust required.

Considering first the drag coefficients of .24 and .334, several factors are evident. The transfer function of $h/-\delta_e$ shows the same character as if the airplane were operated at a speed greater than that for minimum thrust required. For stability, the gain must be greater than Odb at the -180 degree phase crossover point. The pilot may then use a moderate amount of pure gain. By insertion of a lead-lag into his own transfer function, the pilot may use an even larger gain. Considering the gains which might normally be required of the pilot-airplane combination, the pilot should have a very favorable opinion of the response at either of these two values of effective drag.

When the effective drag coefficient is .1345, the response is similar to that of the airplane flying at an airspeed corresponding to that for minimum thrust required. The stability criterion on the Bode plot is the same as in the

previous cases. In order to have a large allowable gain at low frequencies, the pilot would insert a lead-lag in his own transfer function. This case is more difficult for the pilot to control than when he acts as a pure gain, and his opinion of this case will be somewhat less favorable than the preceding cases.

For the basic airplane the drag coefficient is .092, and the stability criterion for the Bode plot as drawn, is that the gain must be 0 db at the +180 degree crossover point. This is due to the presence of the nonminimum phase zero in the transfer function. Since the phase angle is always less than + 180 degrees, there always exists an instability. This is also shown on the root locus, and shows that for any gain there is a pure divergence in altitude. The pilot in this case would be required to use negative gain. This in turn shifts the stability criterion to the 0 db phase angle on the Bode plot. To further improve the response, the pilot will probably insert a lead-lag in his own transfer function. Even with this modification, the pilot will be limited to a lower value of gain than with the preceding cases. Thus in order to satisfactorily fly the airplane under this condition, the pilot will not only be required to insert a lead-lag into the system, but must reverse the polarity of his transfer function. The pilot will not have a very favorable opinion of this condition, but will accept it if necessary to fly the airplane.

When the effective drag coefficient is zero, the root locus shows that there is always an instability present in the transfer function. The pilot may still be able to control the airplane under these conditions, but will be required

to reverse the polarity of his transfer function, and provide a lead-lag. For this condition, the 0 db crossover must occur when the phase angle is above 0 degrees. The lead-lag can bring the phase above this value for some values of gain, but the pilot will be extremely limited in the gain he is allowed to use. This corresponds to a conditionally stable system. Use of reversed polarity by the pilot, a lead-lag, and a limited range of useable gain will make it very difficult for the pilot to control the airplane, and his opinion will probably be in the category of unacceptable for flight.

TRANSIENT RESPONSE

Examination of the transient response of the airplane to a control input may be made either by a computer study, by flight evaluation, or by analytic methods. In the study of the test airplane, all three were used to advantage. The computer study and flight test results are discussed in later sections. In the analytic study, only the results of varying the effective drag were examined. As the root locus in Figure 22 shows, for increasing values of $C_{D_{eff}}$, the phugoid roots become more heavily damped, while the short period mode is virtually unaffected. For three of the previously noted values of $C_{D_{eff}}$ appropriate phugoid and short period roots were obtained. These are shown in Figures 24 through 26. These plots were evaluated for the transient responses by the graphical residue method.

The method of graphical residues may be used quite effectively in such an analysis to gain a rough estimate of the airplane response with a minimum of work. This method allows rapid evaluation of the direction and magnitudes of the responses to impulse, step or ramp inputs. The steady state value is also readily obtained. In this particular analysis the following are at once apparent:

1. The time varying term, present for step input, will be either positive or negative, dependent on the phase of the numerator term.

2. The short period may be ignored in the transient response with negligible error.

The resultant transient responses are shown in Figure 27, plotted together for comparison. Although it will be noted that the initial character of the response is in the same direction, the reversal in the time varying term is evident in the first two cases. As the value of the effective drag is increased, the magnitude of the phugoid oscillation is also noticeably damped. The curves show no obvious error due to omission of the short period mode. Further analysis of the nature of the curves show that due to an impulse type of disturbance, the correction to the new steady state condition will be slow and continuous due to light phugoid damping in the basic airplane. This oscillation is reduced with the improved damping at larger values of $C_{D_{eff}}$. The net result is that the oscillations will damp out much more rapidly at the larger values of $C_{D_{eff}}$.

While it is obvious that the motions under investigation are relatively long term motions which the pilot can easily control by means of the various feedbacks previously discussed, it is highly probable that rather than make continuous control motions during the approach, in order to follow a glide slope, or maintain altitude, that with proper control of the effective drag with feedback quantities to an automatic throttle, that he may exercise more precise control with a minimum of control motion. The net result then hopefully will be an effective stabilization of airspeed and exact altitude control.

COMPUTER STUDY

The computer study was made in order to verify the theoretical results obtained, and to predict flight responses of the airplane to step elevator inputs. The mechanics of the computer setup are presented in Appendix D. The response of the dynamic model of the airplane to step inputs of elevator, with and without elevator feedback, was investigated at the values of $C_{D_{eff}}$ used in the theoretical analysis. The responses at various values of $C_{D_{eff}}$ without elevator position feedback show what amounts to an almost exact correspondence with the analytic solutions. The comparison may be made by comparing the various curves of Figure 27 with the altitude responses of Figures 28 through 32. The increased damping with increased $C_{D_{eff}}$ is quite evident. The steady state velocity in each case approaches the same value. However, the steady state pitch angle

is approximately a linear function of the effective drag. This is explained as follows: The final velocity is equal for all cases, and therefore the steady state angle of attack is equal in each case. However, as the value of effective drag increases, the thrust is decreased, and in order to reach the same velocity, the flight path angle must be steeper. Since the angles of attack are equal, the pitch angle will reach an increasingly higher steady state value for increasing values of effective drag. There is then a direct correspondence on the figures noted between the rate of change of altitude and the steady state pitch angle.

The reversal of the sign of the steady state altitude response to the step input is also evident. In the cases corresponding to operation on the back side of the thrust required curve, the direction of response is the opposite of that for normal flight conditions (front side of the thrust required curve).

The initial response in all cases is almost identical. This response is in the desired direction in all cases, but under the conditions where the transfer function numerator has a nonminimum term, the increase in airspeed due to the initial response results in a decreased drag.

In order to cancel the factors causing the reversal in the direction of the steady state altitude response, the application of an elevator position feedback to the throttle was investigated on the computer. The theory for this method of control has previously been discussed. The results of this study on the computer are presented in Figures 33 through 37. These traces show two important results:

1. The reversal of the sign of the steady state altitude response is eliminated, and the response is in the proper direction regardless of whether the flight condition represents operation on the front or back side of the thrust required curve.

2. The magnitude of the initial phugoid motion is reduced, resulting in less deviation in altitude and airspeed.

ALTITUDE CONTROL AND THE THRUST REQUIRED CURVE

The ease with which the altitude and airspeed can be controlled is a function of the aircraft position on the thrust required curve. Figure 38 is a typical thrust required curve.

The thrust required, T_R, curve is determined by the aircraft drag and is fixed. The thrust available curve, T_A, is adjusted by the pilot within the limitations of the engine. When $T_A = T_R$ there is no net energy input into the system and the aircraft will maintain a constant energy level. This is the case for an aircraft in level flight at a constant airspeed. If however, $T_A \neq T_R$ then the energy level will undergo a change. If T_A is greater than T_R the aircraft must climb or accelerate. If smaller, the aircraft must decelerate or descend. This energy relation is valid no matter what the aircraft speed.

At any given altitude the pilot has no control over the T_R curve, but may move along it by changing speeds. The T_A curve, on the other hand, may be altered by the pilot as mentioned above.

Consider operation on the T_R curve at point 1 in Figure 38. Here the aircraft is on the "front side" of the thrust required curve. The aircraft is at a stable speed condition. If the speed is disturbed it will tend to return to the original speed. Suppose the speed is reduced to V_{1a} by a disturbance, then T_A will exceed T_R and the aircraft will accelerate. Similarly if the disturbance causes the speed to become V_{1b} then T_R exceeds T_A and the aircraft will decelerate.

This speed stability does not hold on the "back side" of the thrust required curve. When on the back side a decrease in speed will cause a T_R greater than T_A and the aircraft will decelerate. Similarly an increase in airspeed will cause the aircraft to accelerate.

Examination of the T_R and T_A curves in Figure 38 also show what the pilot must do to change airspeed or altitude.

To accelerate or climb T_A must be greater than T_R. This can be done by increasing T_A (adding throttle) or by reducing T_R (reducing the drag).

Consider an aircraft at point 1 in Figure 38. If the pilot wishes to climb he may add throttle, thereby producing an excess thrust, and climb at the same airspeed, or he may reduce the required thrust by reducing airspeed and climb at the slower airspeed. The pilot may also establish a faster airspeed by adding throttle. In fact, he must add throttle to increase airspeed at the same altitude.

If the aircraft is on the back side of the curve, at point 2, the pilot may again climb by increasing T_A or decreasing T_R. In this case the pilot must increase airspeed to decrease T_R. This requires the unnatural action of nosing over to climb. This is the effect of the sign reversal of the $h/-\delta_e$ transfer function mentioned previously, which requires the pilot to switch polarity. To increase speed, the pilot, when on the back side, can reduce T_R by nosing over and can then increase speed from V_2 to V_1 at the same altitude. To fly at decreased speed now however the airplane will require additional thrust to maintain level flight.

At the bottom of the thrust required curve the pilot must add throttle when increasing or decreasing speed at the same altitude. He may descend at the same T_A at either a higher or a lower airspeed. To climb however, T_A must be increased.

That the longitudinal control of the aircraft is much more difficult on the back side of the thrust required curve is readily apparent from the above discussion. An appreciation of the relationship between the thrust required curve, thrust available curve, and the longitudinal performance is necessary to understand the problem which faced the pilots who gave their opinions on the tasks they were assigned in this investigation.

The tasks assigned were:

1. Maintain altitude and airspeed.
2. Descend 50 ft. and hold the new altitude and the original airspeed.
3. Establish and hold a 200 ft./min. rate of descent at the same airspeed.
4. Maintain altitude without throttle control.
5. Descend 100 ft. and hold the new altitude without throttle control.

These tasks were performed at the same airspeed, but the velocity feedback to the throttle effectively changed the thrust required curve.

Task number one, maintaining altitude and airspeed is very easy when on the front side of the curve. As mentioned before, the speed is stable. Altitude can be maintained by throttle, adjusting T_A to return to the desired altitude, or by elevator, changing T_R so as to return to the original altitude. On the back side, this becomes more difficult. Speed is no longer stable. The pilot must use the elevator control to maintain airspeed. The altitude can be maintained by elevator but the direction of elevator motion is changed. Unlike the front side, the initial transient response when on the back side and using reversed command is away from the desired altitude. The pilot does much better if he

maintains altitude with the throttle control. In such a case the aircraft response is of the same character as when on the front side.

Task number two, a fifty foot altitude change and then holding the altitude at the original airspeed, is the type maneuver a pilot would make when he became aware of an altitude error. This maneuver can be performed by throttle control or elevator control. When performed with throttle control the position of the aircraft on the thrust required curve does not matter for the T_A is merely made less and the airspeed held constant. When the new altitude is reached T_A is returned to its original value and the transition is complete. The speed of the transition to the new altitude is determined by the thrust reduction. This is the method recommended for all type approaches.

When on the front side of the curve this altitude change can also be easily accomplished without using the throttle. In this case the aircraft is nosed over and gains speed which increases T_R. At the new altitude the aircraft is leveled. The excess speed is diminished as the aircraft dissipates energy due to the excess thrust required. The new altitude can be reached quickly, however there will be some delay before the airspeed is stabilized again.

This maneuver is more difficult to perform with elevator only on the back side, however it can be done. Suppose first though, that the same technique is used as when on the front side. In this case when the new altitude is

reached the excess speed will reduce the thrust required and the aircraft will accelerate to the front side of the curve where it will stabilize. This technique is obviously not suitable. The pilot must then, use the reverse command to the elevator. To lose altitude he must decrease airspeed by pulling back on the stick. He should maintain the new airspeed until he approaches the new altitude. At the slower airspeed T_R exceeds T_A and the aircraft must descend. As the new airspeed is approached the pilot must nose over to pick up speed and if his airman's eye is good he may be at the right altitude and the initial speed at the same time. Such a maneuver is very difficult and is inherently slow as the initial speed reduction must be small so as not to stall the aircraft and the initial transient response is away from the desired correction.

Task number three, to establish and maintain a rate of descent is very similar to task number two. The rate of descent may be established by decreasing the thrust available and maintaining airspeed or by altering the airspeed (hence the thrust required) and maintaining the thrust available. If the throttle is used to establish the rate of descent the response is similar for airspeeds on the front or back side of the thrust required curve. The rate of descent is adjusted by the amount of reduction in throttle. If the rate of descent is established by increasing the thrust required then the airspeed must be increased when on the front side and decreased when on the back side. The rate of descent is dependent on the amount of increase or decrease of airspeed. On

the back side the pilot has less airspeed to play with and can not establish as high a rate of descent. Also on the back side the initial transient response necessary to establish a rate of descent is a gain in altitude. The response then, on the back side is slower than on the front when only the elevator is used.

Task number four, to maintain altitude without throttle control was covered in the discussion concerning task number one.

Turns add to the difficulty in maintaining altitude, however analysis of the thrust required curve reveal that even on the back side the altitude can be maintained in a turn without using the throttle. This is done by increasing the speed on entering a turn and holding the increased speed during the turn.

Task number five was covered fully in the discussion under task number two.

From the above it should be obvious that although the altitude can be controlled with the elevator at all flight speeds it is extremely difficult when below the minimum thrust required speed.

The most foolproof rule appears to be: Control airspeed with the elevator and altitude with the throttle. Not only does this give responses which do not change in character with changes in airspeed, but also the responses in general are more rapid.

EQUIPMENT

The North American Navion (Figure 39) in which the investigation was conducted had previously been equipped with an autopilot and associated actuators for automatic actuation of the rudder, elevator and ailerons. This installation is discussed fully in Reference 2. Several modifications and additions to the existing system were necessary for this study.

A throttle linkage arrangement was installed, which worked in parallel with the existing linkage of the manual throttle. This arrangement was chosen as it allowed shifting from the automatic to manual throttle operation with no repositioning of the manual system. The cable was routed around the engine from the carburetor, and terminated at the throttle servo just forward of the pilot's seat on the cockpit flooring. This position allowed checking of the actuator operation and also allowed manual disconnection of the linkage in an emergency situation. The cable was attached to the servo actuating arm by a wing nut to allow easy removal. A rotary actuator was chosen rather than a linear actuator for several reasons. First, the rotary actuator was more easily adapted to the parallel system used than would have been the case with a linear actuator. The type of rotary actuator used was also compatible with the autopilot system in the airplane, and was readily available for installation. The use of this actuator introduced a certain degree of nonlinearity into the throttle operation, however, the movement about the desired trim speed was

sufficiently small as to approximate a linear condition. The actuator arm was one and one-half inches radius, and drove the throttle through its full linear travel with approximately 60 degrees of angular motion. Therefore for the small throttle adjustments about trim speed, this was nearly linear.

In addition to the servo and throttle linkage, an electric, hand operated throttle quadrant was installed to enable the pilot to make large throttle changes for trim readjustment and maneuvering with the autopilot engaged. This throttle and the servo are shown in Figure 40.

For velocity changes of the order encountered in the normal phugoid motion, the normal pilot-static pressure probe did not provide sufficient differential pressure to actuate the associated pressure transducer. The pressure transducer utilized had a full scale deflection of 0 to +1.5 psia, and 30 volts applied. To increase the system resolution, a Venturi Tube was arranged on a boom approximately one-half chord length ahead of the right wing, for the purpose of reducing the static pressure. Mounted on the boom above the Venturi Tube was the total head probe with an enclosing shield oriented parallel to the Venturi Tube. Utilizing this arrangement, an effective amplification of the pressure differential of approximately 8 was obtained. The installation as installed on the airplane is shown in Figure 41.

CALIBRATION

The initial flight was performed to check the velocity feedback to the throttle system. The apparent shift of the thrust required curve, with greater velocity feedback, was observed by using the saw tooth method for determining the thrust required curve at different feedback gains.

The second flight consisted of an examination of the aircraft responses to step elevator, with different feedback gains. The results of this are shown in Figure 42. From this data the calibration of $C_{D_{eff}}$ for feedback gains can be estimated by the procedure below.

The basic airplane, with no velocity feedback, has a C_D of .092. This is verified by a check of the phugoid damping and was used as a basis for evaluating the other responses.

The analysis of the response to step elevator inputs by the residue method showed that the rate of climb after the phugoid is damped is proportional to $(C_L C_{D\alpha} / C_{L\alpha}) - C_{D_{eff}}$. For the Navion at the test conditions used

$C_L C_{D\alpha} / C_{L\alpha} = .1345.$

Therefore $K\ (R/C) = .1345 - C_{D_{eff}}$.

K, the proportionality factor, is found from the curve for the basic airplane, where $C_{D_{eff}} = .092$. Using this K the effective drag coefficients at other gain settings were found. Figure 43 is the calibration curve.

PROCEDURE USED IN OBTAINING PILOT OPINIONS

Pilots were asked to give their opinions on the five tasks mentioned before. These were

1. Maintain altitude and airspeed.
2. Descend 50 ft. and maintain the new altitude at the initial airspeed.
3. Establish and hold a 200 ft./min. rate of descent.
4. Maintain altitude without throttle control.
5. Descend 100 ft. and maintain the new altitude without throttle control.

The pilots performed these tasks at the same airspeed (80 mph) but at different values of $C_{D_{eff}}$. These different values of $C_{D_{eff}}$ change the apparent position of the aircraft on the thrust required curve.

A numerical rating system was used. This system is shown in Table IV.

Three pilots performed the flight evaluation program. All three were Navy pilots of varied background whose experience levels averaged in excess of 2000 hours flight time.

RESULTS

The results of the flight evaluation phase of the investigation are presented in Figures 44 and 45. The five curves of these figures represent a typical

sampling of the various types of maneuvers in the longitudinal mode which a pilot will be likely to perform in a landing approach.

The character of the curves is such that a trend toward improved pilot opinion is evident as the value of $C_{D_{eff}}$ is increased. All curves tend to become asymptotic to a pilot opinion value of 2, at high gain. The degree of scatter is also reduced at higher values of $C_{D_{eff}}$, showing that the bandwidth of pilot opinion is reduced at high gain.

Verbal comments were also obtained from each pilot in order to clarify the results of the curves. A general statement summarizing each pilot's comments follows:

Pilot 1. The improvement in the ability to maintain altitude with a reduced workload was obvious. The ability to make turns while on the back side of the curve was greatly improved. Further sophistication in the system would be desirable.

Pilot 2. The automatic throttle feature was of some aid in flying on the back side of the curve, but did not change the response very much except at high gain.

Pilot 3. The response showed virtually no change at low gain. Such a system would give favorable results if sufficient gain were utilized.

DISCUSSION OF RESULTS

The results obtained from the flight test show that the pilot prefers to fly the airplane on the front side of the thrust required curve for the conditions tested. In all except one case, the pilot opinion improved markedly when flying at a condition simulating operation on the front side of the curve. There were a number of factors which apparently had an effect on pilot opinion.

To begin with, the pilot did not like to have the reversal of steady state response associated with operation on the back side of the curve. He preferred to use forward stick movement to decrease altitude. The back side operating condition, where the initial response is in the proper direction, followed by a reversal with time, is not a natural condition to the pilot. As the condition of no feedback ($C_{D_{eff}}$ of .092) indicates, the pilot did not have a favorable opinion of operation near minimum airspeed, and the subsequent requirement of inserting a lead term in his transfer function to maintain control. The conditions of increased gain show that this unfavorable opinion was no longer existent when the airplane was operating on the front side of the curve, even though the airspeed had not been changed. In addition, the lightly damped phugoid also gave the pilot considerable trouble. When he had adapted himself to supply the necessary lead term, he encountered the phugoid oscillation. This initially appeared to him as a response in the proper direction, but as he

attempted to exercise exact control, he inevitably corrected in the wrong direction, probably because the operating point was on the back side of the curve. The pilot also had more difficulty in controlling the airplane in conditions of unbalanced flight. This was due to the different thrust requirements as the degree of rudder uncoordination varied.

Another factor which had a bearing on the pilots opinion was the ability to maintain an exact airspeed. In smooth air when it was possible to hold the speed constant, the throttle feedback, working on airspeed variation, was not activated. This resulted in a tendency for the pilot opinion to be relatively constant regardless of gain. In turbulent air when it was not possible to hold constant airspeed, the pilot opinion improved noticeably with gain. This system is obviously of no benefit when the pilot is able to maintain an exact airspeed in his approach. However, the conditions are rarely such that this is possible for the pilot.

The rate of transition from one flight condition to another, i.e., level to descending or descending to level, made a noticeable difference in pilot opinion. Considering the basic airplane with no feedback, with a rapid transition, the tendency was to enter into a phugoid type oscillation about the new flight condition. The effect of throttle feedback, where the effective drag was increased, was to nullify this and allow the pilot to make more rapid transitions with a greater degree of precision.

The condition where the effective drag of the airplane was decreased to $C_{D_{eff}} = 0$, clearly showed the unstable phugoid motion and divergence from the initial conditions due to any disturbance.

In addition to the general remarks, each curve of the results showed some measure of information about the pilots reaction and opinion.

Condition 1. In this instance, the pilot attempted to maintain control by use of both throttle and elevator. At conditions representing operation on the back side of the curve, the pilot experienced difficulty in obtaining a proper coordination of elevator and throttle. This was a result of attempting to maintain altitude with throttle, using a unity feedback of altitude. As the root locus of this shows (Figure 16), this drives the phugoid unstable. Reference 3 shows a similar condition very vividly. In a somewhat related test, a plot of engine thrust versus time shows that on an approach using manual throttle operation, the engine thrust follows the airplane phugoid motion almost exactly. This same condition exists for unity feedback utilizing elevator control. Thus, in order to have a stable system, the pilot was required to provide a lead in his own transfer function to exercise effective control. As the gain of the feedback to the throttle was increased, the burden of throttle control was shifted from the pilot due to the removal of the need for him to supply a lead in the system. The opinion showed a corresponding improvement.

Condition 2. In this instance, the pilot tried to lose 50 feet and then hold the new altitude. Again the difficulty of controlling the altitude with use of unity feedback to either elevator or throttle was evident at zero and low gain values. As the pilot made the transition to lose the altitude, the tendency was to use forward stick movement. The throttle was also reduced. This resulted in an initial motion in the desired direction. When thrust was again added, to restore the initial equilibrium, the steady state due to elevator deflection returned the airplane to the original altitude and in some instances to a higher altitude before corrective action was initiated. Superimposed on this was the phugoid type motion which the pilot attempted to control with elevator, without much success. However, again, as in the situation previously discussed, the pilot opinion improved with increasing $C_{D_{eff}}$. This was apparently the result of both the removal of the nonminimum phase condition and the improved phugoid damping.

Condition 3. Here the pilot was told to establish and hold a 200 ft./min. rate of descent. There was a marked improvement in the pilots ability to establish and hold this steady rate with increased $C_{D_{eff}}$. Increased phugoid damping eliminated the oscillations more rapidly and allowed the pilot to establish the steady rate more easily.

Condition 4. In this portion of the flight test, the pilot attempted to maintain the altitude constant without use of the throttle. In performing this task, the pilot was able to do this satisfactorily once he had established equilibrium conditions. The execution of level turns however, gave considerable trouble, especially at low gain. At high gain, the pilot was able to maintain level flight in coordinated turns, and the higher the gain, the steeper the turn it was possible to hold. The degree of scatter of the points at low gain seem to indicate that this is an unfamiliar maneuver and it is difficult for the pilot to form a definite opinion.

Condition 5. In the final instance, the pilot was to lose 100 feet and maintain the new altitude without use of manual throttle. Once at or above the basic airplane effective drag, there appeared to be no change in the ability of the pilot to perform this task. The most probable explanation for this result lies in the rate at which the altitude was decreased. As previously noted, the effect of maintaining a constant airspeed is to render the velocity feedback to the throttle ineffective. In this portion of the test, the tendency of the pilot was to use only a small elevator deflection, and thereby hold the airspeed change to a low value. This allowed a slow change of altitude which was easily controlled by the pilot. As the gain was increased, the rate of loss of altitude was increased, but the opinion was essentially the same. It is

apparent that as larger increments of altitude change are made, the rate of this change has a bearing on the pilot opinion. This tends to have some correlation with the previously mentioned fact that changes made on the back side of the curve tend to be at a slower rate than those at points on the front side of the curve.

CONCLUSIONS

From this investigation the following is concluded:

1) Altitude control at airspeeds below the minimum thrust required speed is difficult. This difficulty is caused by:
    a) An unfavorable altitude response to elevator which requires an initial response away from the desired altitude.
    b) The need for a greater number of control motions to effect transitions.
    c) The phugoid mode tends to be less damped and is more easily excited.
2) The autopilot controlled throttle is beneficial to altitude control at speeds below the minimum thrust required speed.
3) Excitation of the throttle by either angle of attack, velocity, or elevator position tends to cause an aircraft at speeds below the minimum thrust required speed to respond to the elevator as though it were at a higher airspeed.
4) The autopilot actuated throttle makes possible more rapid transitions at low airspeeds.
5) The degree of benefit derived from the velocity controlled throttle servo is dependent on the pilot technique and his ability to hold airspeed. The poorer his technique or his ability to hold airspeed the greater the advantage he would derive.

6) At speeds above the minimum thrust required airspeed, little improvement in altitude control is achieved by use of a velocity actuated throttle servo.

7) The pilots choice of approach airspeed is affected by altitude control.

8) The best technique in flying an approach is to control airspeed with elevator and altitude with throttle.

RECOMMENDATIONS

It is recommended that:

1) Further investigation be made to determine
    a) The relative importance of phugoid damping versus position on the thrust required curve.
    b) The influence of lateral stability on the landing approach.
2) A flight investigation be conducted using velocity, angle of attack and elevator position feedback to the throttle, to determine an optimum system for airplane control during the landing approach.
3) An investigation of landing approach problems peculiar to high performance airplanes be conducted utilizing a flight simulator.
4) A rigid throttle linkage be installed in the test airplane prior to further testing.

REFERENCES

1. Perkins, C. D., and Hage, R. E.; Airplane Performance, Stability and Control. John Wiley and Sons, Inc., New York, 1958.

2. Zink, S. T., and Wiltsie, R. E.; Flight Simulation of the Longitudinal Motions of Another Aircraft. Aeronautical Engineering Report No. 508, Princeton University, May 1960.

3. Lina, L. J. et al.; Flight Investigation of an Automatic Throttle Control in Landing Approaches. NASA Memorandum 2-19-596, March 1959.

4. Crigler, J. L., and Jaquis, R. E.; Propeller Efficiency Charts for Light Aircraft. NACA TN 1338, July 1947.

APPENDIX A

STABILITY DERIVATIVES

The airplane stability derivatives were calculated for the North American Navion No. N91566, at an altitude of 4500 feet s.d.a., with the airplane in the following configuration:

Landing gear and flaps ----- UP
Cowl flaps ----- FAIRED
Canopy ----- CLOSED
Propeller ----- 2000 rpm
V_{t_o} (level flight) ----- 80 mph
ρ_o ----- .00208 slugs/ft.^{3}

From Reference , the following stability derivatives were obtained:

C_{L_α} = 5.45/radian

$C_{m_{\delta_e}}$ = -p.435/radian

C_{m_u} = zero (assumed)

$$W/S = \frac{2750}{184.2} = 14.93 \text{ lb/ft.}^3$$

Under the assumption that the installations necessary for the automatic throttle would have a negligible effect on the c.g. location, the last previously determined value of C_{m_α} as given by Reference 2 was also taken to be the correct value, i.e., $C_{m_\alpha} = -.485$/radian.

CALCULATIONS

$$q = \tfrac{1}{2}\rho V^2 = \frac{.00208}{2}(117.3)^2 = 14.31 \text{ lb/ft}^2$$

$$C_{L_0} = \frac{W/S}{q} = \frac{14.93}{14.31} = 1.04$$

$$C_{D_0} = C_{D_f} + \frac{C_L^2}{\pi A e} = .025 + \frac{(1.04)^2}{\pi(6.04)(.85)} = .092$$

$$C_{D_\alpha} = \frac{2 C_{L_0} C_{L_\alpha}}{\pi A e} = \frac{2(1.04)(5.45)}{\pi(6.04)(.85)} = .705$$

$$\tau = \frac{m}{\rho S V} = \frac{2750/32.2}{.00208(184.2)(117.3)} = 1.895$$

$$\bar{h} = \frac{2(k_y)^2}{\mu \tau^2} = \frac{2(33.8)}{\frac{2750/32.2\,(5.7)^2}{.00208(184.2)(5.7)}} = .0532$$

From Reference 2, the following values were taken and a correction applied for the difference in test altitude:

$$C_{m_{d\alpha}} = -.0745/\text{radian}$$

$$C_{m_{d\theta}} = -.1595/\text{radian}$$

The following value was determined as shown in Appendix E.

$$C_{T_{\delta_t}} = .0895 \text{ in.}^{-1}$$

APPENDIX B

DEVELOPMENT OF ALTITUDE TO CONTROL MOVEMENT TRANSFER FUNCTIONS

CHARACTERISTIC EQUATION

Utilizing the nondimensional equations of motion as developed in Reference 1,

$$\begin{aligned}
&(C_D + d)u &&+ \tfrac{1}{2}(C_{D_\alpha} - C_L)\alpha &&+ \tfrac{1}{2}C_L\theta &&= 0\\
&C_L u &&+ \left(\tfrac{C_{L\alpha}}{2} + d\right)\alpha &&- d\theta &&= 0\\
&0 &&+ (C_{m_\alpha} + C_{m_{d\alpha}} d)\alpha &&+ (C_{m_{d\theta}} d - \bar{h} d^2)\theta &&= 0
\end{aligned}$$

Solution of the determinant, and expressing the solution in Laplace notation,

$$\Delta(s) = C_4 s^4 + C_3 s^3 + C_2 s^2 + C_1 s + C_0$$

where

$$C_4 = -\bar{h}$$

$$C_3 = C_{m_{d\theta}} + C_{m_{d\alpha}} - \bar{h}\left(C_D + \frac{C_{L\alpha}}{2}\right)$$

$$C_2 = C_{m_\alpha} + \frac{C_{L\alpha} C_{m_{d\theta}}}{2} + C_D(C_{m_{d\theta}} + C_{m_{d\alpha}}) + \frac{\bar{h}}{2}(C_L C_{D_\alpha} - C_D C_L - C_L^2)$$

$$C_1 = C_D C_{m_\alpha} + C_L^2(C_{m_{d\theta}} + C_{m_{d\alpha}})/2 + C_{m_{d\theta}}(C_D C_{L_\alpha} - C_L C_{D_\alpha})/2$$

$$C_0 = C_L^2 C_{m_\alpha}/2$$

Dividing through the coefficients by $-\bar{h}$ to obtain a unity coefficient for the C_4 term, and solving the determinant with C_D as a variable,

$$s^4 + (C_D + 7.11)s^3 + (7.12\,C_D + 17.443)s^2 + (17.29\,C_D + 1.277)s + 4.93 = 0$$

Substitution of the normal Navion drag coefficient of .092 yielded the following quartic:

$$\frac{\Delta(s)}{-\bar{h}} = s^4 + 7.20\,s^3 + 18.1\,s^2 + 2.87\,s + 4.93$$

This quartic yielded the following quadratic factors when solved by Lin's method:

$$(s^2 + .048\,s + .282)\,(s^2 + 7.152\,s + 17.47)$$

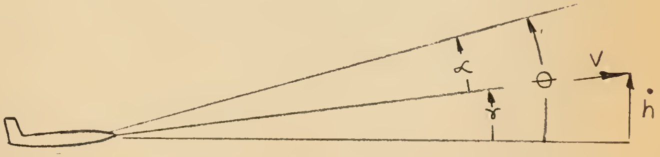
These terms represent the roots of the characteristic equation of the longitudinal modes of motion. The first term is the phugoid mode, and the second term represents the short period mode.

The frequencies and damping of the roots are:

	frequency	damping
phugoid	.532	.024
short period	4.18	.853

NUMERATOR

In the development of the numerator of the transfer function, the relationship shown in the following figure was utilized as the starting point.



$$\gamma = \theta - \alpha$$

From the figure, it will be noted that the vertical velocity $\dot{h}$ is given by $V \sin \gamma$. When limited to small angles, this is approximately equal to $V\gamma$. Integrating the rate of change of altitude, altitude is obtained or

$$h = \frac{V\gamma}{S} \qquad \text{and on substituting for } \gamma$$

$$h = \frac{V(\theta - \alpha)}{S}$$

Solution of the determinants for the applied elevator forcing function yielded

$$\theta(S) = \frac{-C_{m_{\delta_e}}\delta_e}{\Delta(S)}\left[S^2 + \left(\frac{C_{L_\alpha}}{2} + C_D\right)S + \frac{C_D C_{L_\alpha}}{2} - \frac{C_L C_{D_\alpha}}{2} + \frac{C_L^2}{2}\right]$$

and similarly

$$\alpha(S) = \frac{-C_{m_{\delta_e}}\delta_e}{\Delta(S)}\left[S^2 + C_D S + \frac{C_L^2}{2}\right]$$

and when combined

$$[\theta - \alpha](S) = \frac{-C_{m_{\delta_e}}\delta_e C_{L_\alpha}}{\Delta(S)}\left[S + C_D - \frac{C_L C_{D_\alpha}}{C_{L_\alpha}}\right]$$

Substitution in the previously derived relationship yielded

$$\frac{h}{-\delta_e} = \frac{-V(\theta - \alpha)}{S\,\Delta(S)} = \frac{V C_{m_{\delta_e}} C_{L_\alpha}}{S\,\Delta(S)}\left[S + C_D - \frac{C_L C_{D_\alpha}}{C_{L_\alpha}}\right]$$

Substitution of the appropriate values for the Navion stability derivatives under the test conditions gave

$$\frac{h}{-\delta_e} = \frac{8650\,(S - .0425)}{S(S^2 + .048S + .282)(S^2 + 7.152S + 17\,47)}$$

TRANSFER FUNCTION AS MODIFIED BY ELEVATOR FEEDBACK TO THE THROTTLE

In order to change the initial response of the airplane to step elevator inputs, the elevator position was fed back to the throttle. This resulted in modification of the drag equation as follows:

$$\left(C_D - \frac{\partial T/\partial V}{\rho S V} + d\right)u + \tfrac{1}{2}(C_{D_\alpha} - C_L)\alpha + \tfrac{1}{2}C_L\theta = C_{T_{\delta_e}}\delta_e$$

where $C_{T_{\delta_e}}$ is the nondimensional coefficient of thrust with respect to elevator displacement. Solution of the equations of motion with elevator feedback then yielded a modified transfer function $h/-\delta_e{}^{*}$, where the asterisk denotes the transfer function with elevator feedback, i.e.,

$$\left(\frac{h}{-\delta_e}\right)^{\star} = \frac{h}{-\delta_e} + \frac{N}{-\Delta(s)}$$

where $h/-\delta_e$ represents the original transfer function with $C_{T_{\delta_e}} = 0$. Solution of the equations of motion for $N/_{-\Delta(s)}$ yielded

$$\frac{N}{-\Delta(s)} = \frac{V C_{T_{\delta_e}} C_L \bar{h}}{s\,\Delta(s)} \left[s^2 + \frac{(C_{m_{d\alpha}} - C_{m_{d\theta}})}{\bar{h}} s + \frac{C_{m_\alpha}}{\bar{h}} \right]$$

Thus $$\left(\frac{h}{-\delta_e}\right)^{\star} = \frac{8650(s-.0425) + 122\,C_{T_{\delta_e}}(s+3.92)(s-2.32)}{s(s^2+.048s+.282)(s^2+7.152s+17.47)}$$

ALTITUDE TO THROTTLE TRANSFER FUNCTION

In examining the effect of throttle variation on altitude, the initial step was examining the effect on the equations of motion. The only change noted was the addition of a throttle effectiveness term to the drag equation to give

$$((C_{D_{eff}} + d)u + \tfrac{1}{2}(C_{D_\alpha} - C_L)\alpha + \tfrac{1}{2} C_L \theta = C_{T_{\delta_t}} \delta_t$$

Initially, h was determined to be $\frac{V(\theta-\alpha)}{s}$, $h/\delta_t = \frac{V(\theta-\alpha)}{s\,\delta_t}$

$$\theta/\delta_t = \frac{C_{T_{\delta_t}}}{\Delta(s)}[C_L((C_{m_\alpha} + C_{m_{d\alpha}} s)]$$

$$\alpha/\delta_t = \frac{C_{T_{\delta_t}}}{\Delta(s)}[C_L((C_{m_{d\theta}} s - \bar{h} s^2)]$$

$$h/\delta_t = \frac{V C_{T_{\delta_t}}}{\Delta(s)}[C_{m_\alpha} + (C_{m_{d\alpha}} + C_{m_{d\theta}})s - \bar{h} s^2]$$

and using proper stability derivative values,

$$h/\delta_t = \frac{122\,C_{T_{\delta_t}}}{\Delta(s)}[s^2 + 4.4s + 9.12]$$

APPENDIX C

ROOT LOCUS FOR VARYING $C_{D_{eff}}$

With varying $C_{D_{eff}}$, the locus of roots of the characteristic equation can be written in the following form:

$$C_4' s^4 + (C_3' + A C_D) s^3 + (C_2' + B C_D) s^2 + (C_1' + C C_D) s + C_0' = 0$$

where

$$C_4' = -\bar{h}$$

$$C_3' = C_{m_{d\theta}} + C_{m_{d\alpha}} - \bar{h} \frac{C_{L\alpha}}{2}$$

$$A = -\bar{h}$$

$$C_2' = C_{m_\alpha} + \frac{C_{L\alpha} C_{m_{d\theta}}}{2} + \frac{\bar{h}}{2} [C_L C_{D_\alpha} - C_L^2]$$

$$B = C_{m_{d\theta}} + C_{m_{d\alpha}} - \frac{\bar{h}}{2} C_L$$

$$C_1' = \frac{C_L^2}{2} [C_{m_{d\theta}} + C_{m_{d\alpha}}] + \frac{C_{m_{d\theta}}}{2} [- C_L C_{D_\alpha}]$$

$$C = C_{m_\alpha} + \frac{C_{m_{d\theta}}}{2} C_{L\alpha}$$

$$C_0 = \frac{C_L^2}{2} C_{m_\alpha}$$

Substituting values for the various stability derivatives, as obtained in Appendix A,

$$s^4 + (C_D + 7.11) s^3 + (7.12 C_D + 17.443) s^2 + (17.29 C_D + 1.277) s + 4.93 = 0$$

or

$$\frac{C_D s [s^2 + 7.12 s + 17.29]}{s^4 + 7.11 s^3 + 17.443 s^2 + 1.277 s + 4.93} = -1$$

Solving the numerator term for location of the zero yielded

$$S = 0 \quad \text{and} \quad S = 3.56 \pm j\,2.13$$

and similarly solving for the denominator terms yielded the following poles

$$S = +.0213 \pm j\,.53 \quad \text{and} \quad S = -3.58 \pm j\,2.17$$

Also to a very close approximation,

$$\frac{C_D S}{(S - .0213 \pm j.53)} = -1$$

This approximation is valid because the complex poles and zeros in the left hand half plane effectively cancel each other. This further illustrates the fact that the short period takes place at relatively constant velocity.

The root locus showing this variation with varying $C_{D_{eff}}$ is shown in Figure 22. This illustrates that for increasing $C_{D_{eff}}$ the phugoid damping is increased, but the phugoid frequency is relatively unaffected for all except very high values of $C_{D_{eff}}$. Examination of the numerator term of the transfer function shows that the zero moves from the region of nonminimum to minimum phase condition with increasing $C_{D_{eff}}$.

APPENDIX D

ANALOG COMPUTER SETUP

The dynamic model of the Navion test airplane was simulated on the Goodyear GEDA Analog Computer Model GN215-13.

The equations of motion as used in the computer analysis were:

Drag: $\dot{u} = -\frac{1}{2}(C_{D_\alpha} - C_L)\alpha - \frac{1}{2}C_L\theta - C_{D_{eff}}u + C_{T_{\delta_e}}\delta_e$

Lift: $\dot{\alpha} = -\frac{C_{L_\alpha}}{2}\alpha + \dot{\theta} - C_L u$

Moment: $\ddot{\theta} = \frac{C_{m_\alpha}}{\bar{h}}\alpha + \frac{C_{m_{d\alpha}}}{\bar{h}}\dot{\alpha} + \frac{C_{m_{d\theta}}}{\bar{h}}\dot{\theta} + \frac{C_{m_{\delta_e}}}{\bar{h}}\delta_e$

Substituting previously determined values of the stability derivatives into the above equations gave:

$$\dot{u} = .168\alpha - .52\theta - C_{D_{eff}}u + C_{T_{\delta_e}}\delta_e$$

$$\dot{\alpha} = -2.72\alpha + \dot{\theta} - 1.04u$$

$$\ddot{\theta} = -9.1\alpha - 1.4\dot{\alpha} - 3.0\dot{\theta} - 28\delta_e$$

In addition, in order to evaluate the altitude deviation from the desired flight path, an additional string was added to the setup. This involved utilization of the equation $h = \frac{V}{s}(\theta - \alpha) = \frac{117.3}{s}(\theta - \alpha)$.

The integration indicated in the equation was performed by a separate amplifier used as an integrator (see Figure 46).

APPENDIX E

DETERMINATION OF THE THROTTLE CONTROL EFFECTIVENESS OF THE NAVION

The throttle control effectiveness is defined here to be the change in engine thrust, in pounds, divided by the throttle control movement, in inches, required to produce the thrust change at 2000 rpm engine speed, 80 mph true air speed, and 4500 feet density altitude. Knowledge of the throttle control effectiveness was required to design the throttle control servo. This appendix covers the procedure used in finding the throttle control effectiveness.

From the Sea Level Part Throttle and the Full Throttle curves for the Continental Series "E" Engine the part throttle curve for 2000 rpm at 4500 feet was found as shown in Figure 47.

The propeller efficiency was estimated from data contained in Reference 4.

The part throttle thrust horse power was found by combining the brake horsepower part throttle and the propeller efficiency curves. This is shown in Figure 48. The part throttle thrust curve was found from the thrust horsepower and the airspeed.

The throttle control movement, $\Delta \ell$, necessary for changes in manifold pressure was found by flight testing a Navion identical to the test vehicle. The flight test result is shown in Figure 48. The throttle control position, 1, for which the manifold pressure is found is not significant as it is merely a reference distance. From the flight test result and the part throttle thrust curve the throttle control effectiveness was estimated.

As shown in Figure 48, the throttle control effectiveness was estimated to be

$$\Delta T/\Delta l = \partial T/\partial \delta_t = 276 \text{ lbs./in.}$$

This must be divided by qS to be used in the Drag Equation and becomes

$$C_{T\delta_t} = (\partial T/\partial \delta_t)/qS = 0.0895 \text{ in.}^{-1}$$

This parameter is not nondimensional but becomes so when multiplied by throttle control movement, if the movement, δ_t, is measured in inches.

TABLE I

TRANSFER FUNCTIONS

$$\frac{h}{-\delta_e} \qquad \frac{V C_{m_{\delta_e}} C_{L_\alpha}}{s\,\Delta(s)}\left[s + C_D - \frac{C_L C_{D_\alpha}}{C_{L_\alpha}}\right]$$

$$\frac{u}{\delta_e} \qquad \frac{-C_{m_{\delta_e}} C_{D_\alpha}}{2\,\Delta(s)}\left[s + \frac{C_L C_{L_\alpha}}{2 C_{D_\alpha}}\right]$$

$$\frac{\alpha}{-\delta_e} \qquad \frac{C_{m_{\delta_e}}}{\Delta(s)}\left[s^2 + C_D s + \frac{C_L^2}{2}\right]$$

$$\frac{\theta}{-\delta_e} \qquad \frac{C_{m_{\delta_e}}}{\Delta(s)}\left[s^2 + \left(C_D + \frac{C_{L_\alpha}}{2}\right)s + \frac{C_D C_{L_\alpha}}{2} + \frac{C_L}{2}\left(C_L - C_{D_\alpha}\right)\right]$$

$$\frac{\dot{h}}{-\delta_e} \qquad \frac{V C_{m_{\delta_e}} C_{L_\alpha}}{\Delta(s)}\left[s + C_D - \frac{C_L C_{D_\alpha}}{C_{L_\alpha}}\right]$$

$$\frac{h}{\delta_t} \qquad \frac{-\bar{h} V C_L C_{T_{\delta_t}}}{s\,\Delta(s)}\left[s^2 - \left(\frac{C_{m_{d\alpha}} + C_{m_{d\theta}}}{\bar{h}}\right)s - \frac{C_{m_\alpha}}{\bar{h}}\right]$$

$$\frac{u}{\delta_t} \qquad \frac{-\bar{h} C_{T_{\delta_t}}}{\Delta(s)}\left[s\left(s^2 + \left\{\frac{C_{L_\alpha}}{2} - \frac{C_{m_{d\theta}}}{\bar{h}} - \frac{C_{m_{d\alpha}}}{\bar{h}}\right\}s - \frac{C_{m_\alpha}}{\bar{h}} - \frac{C_L C_{m_{d\theta}}}{\bar{h}}\right)\right]$$

$$\frac{\alpha}{\delta_t} \qquad \frac{-C_{T_{\delta_t}}}{\Delta(s)}\left[C_L\left(C_{m_{d\theta}} s - \bar{h} s^2\right)\right]$$

$$\frac{\theta}{\delta_t} \qquad \frac{C_{T_{\delta_t}} C_{m_{d\alpha}}}{\Delta(s)}\left(s + \frac{C_{m_\alpha}}{C_{m_{d\alpha}}}\right)$$

$$\frac{\dot{h}}{\delta_t} \qquad \frac{-\bar{h} V C_L C_{T_{\delta_t}}}{\Delta(s)}\left[s^2 - \left(\frac{C_{m_{d\alpha}} + C_{m_{d\theta}}}{\bar{h}}\right)s - \frac{C_{m_\alpha}}{\bar{h}}\right]$$

TABLE II

STABILITY OF THE LONGITUDINAL TRANSFER FUNCTIONS WITH UNITY FEEDBACKS

TRANSFER FUNCTION	REMARKS
U/δ_e	PHUGOID BECOMES UNSTABLE AT LOW GAIN
$\alpha/-\delta_e$	STABLE FOR ALL GAIN
$\theta/-\delta_e$	STABLE FOR ALL GAIN
$\dot{h}/-\delta_e$	STABLE BUT FOR VERY HIGH GAIN
$h/-\delta_e$	UNSTABLE WHEN $V < V_{\delta_{t_{min}}}$
U/δ_t	STABLE FOR ALL GAIN
α/δ_t	UNSTABLE BUT AT LOW GAIN
θ/δ_t	UNSTABLE BUT AT LOW GAIN
$\dot{h}/\delta_t$	STABLE FOR ALL GAIN
h/δ_e	PHUGOID BECOMES UNSTABLE AT LOW GAIN

TABLE III

STEADY STATE RESPONSE TO STEP INPUT

TRANSFER FUNCTION	MAGNITUDE PROPORTIONAL TO	EFFECT OF $+\Delta V$	REMARKS
$U/-\delta_e$	$C_{m_{\delta_e}} C_L C_{L_\alpha} / \bar{h}$	less -	Is always negative
$\alpha/-\delta_e$	$C_{m_{\delta_e}} C_L^2 / \bar{h}$	less +	Is always positive
$\theta/-\delta_e$	$C_{m_{\delta_e}} [C_D C_{L_\alpha} + C_L (C_L - C_{D_\alpha})] / \bar{h}$	less +	Is always positive
$\dot{h}/-\delta_e$	$C_{m_{\delta_e}} (C_D C_{L_\alpha} - C_L C_{D_\alpha}) / \bar{h}$	more +	0 at $(\delta_t)_{min}$
$h/-\delta_e$	$\pm \infty$		$h = \int \dot{h} dt$
U/δ_t			Always 0
α/δ_t	$C_{T_{\delta_t}} C_L$	less +	Is always positive
θ/δ_t	$C_{T_{\delta_t}} C_{m_\alpha} / \bar{h}$	less +	Is always positive
$\dot{h}/\delta_t$	$V C_L C_{T_{\delta_t}} C_{m_\alpha}$	less +	Is always positive
h/δ_t	$+ \infty$		$h = \int \dot{h} dt$

TABLE IV

RATING SCALE

NUMERICAL RATING	CATEGORY	ADJECTIVE DESCRIPTION WITHIN CATEGORY
1	ACCEPTABLE AND SATISFACTORY	EXCELLENT
2		GOOD
3		FAIR
4	ACCEPTABLE BUT UNSATISFACTORY	FAIR
5		POOR
6		BAD
7	UNACCEPTABLE	BAD
8		VERY BAD
9		DANGEROUS
10	UNFLYABLE	

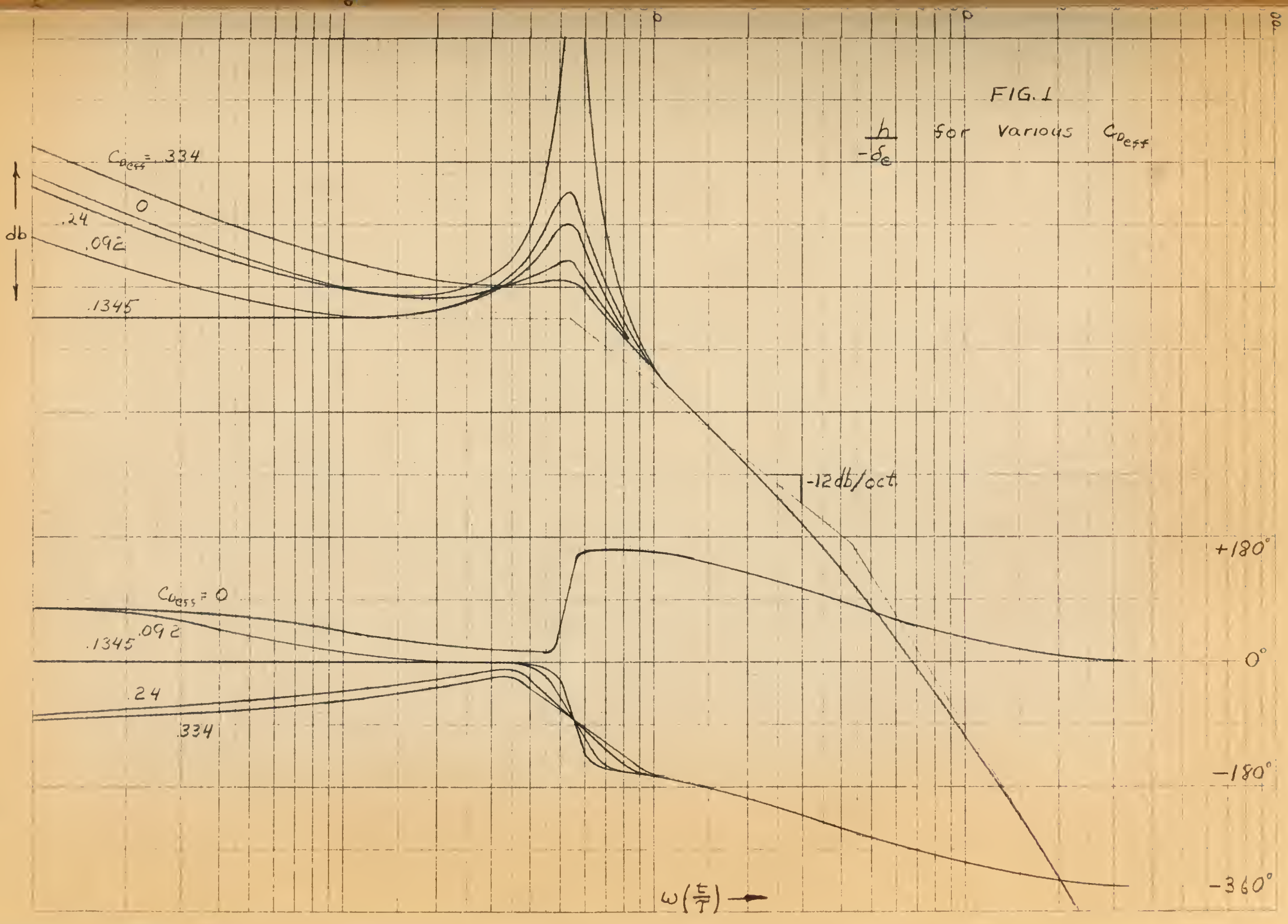


FIG. 1

$\frac{h}{-\delta_e}$ for various $C_{D_{eff}}$

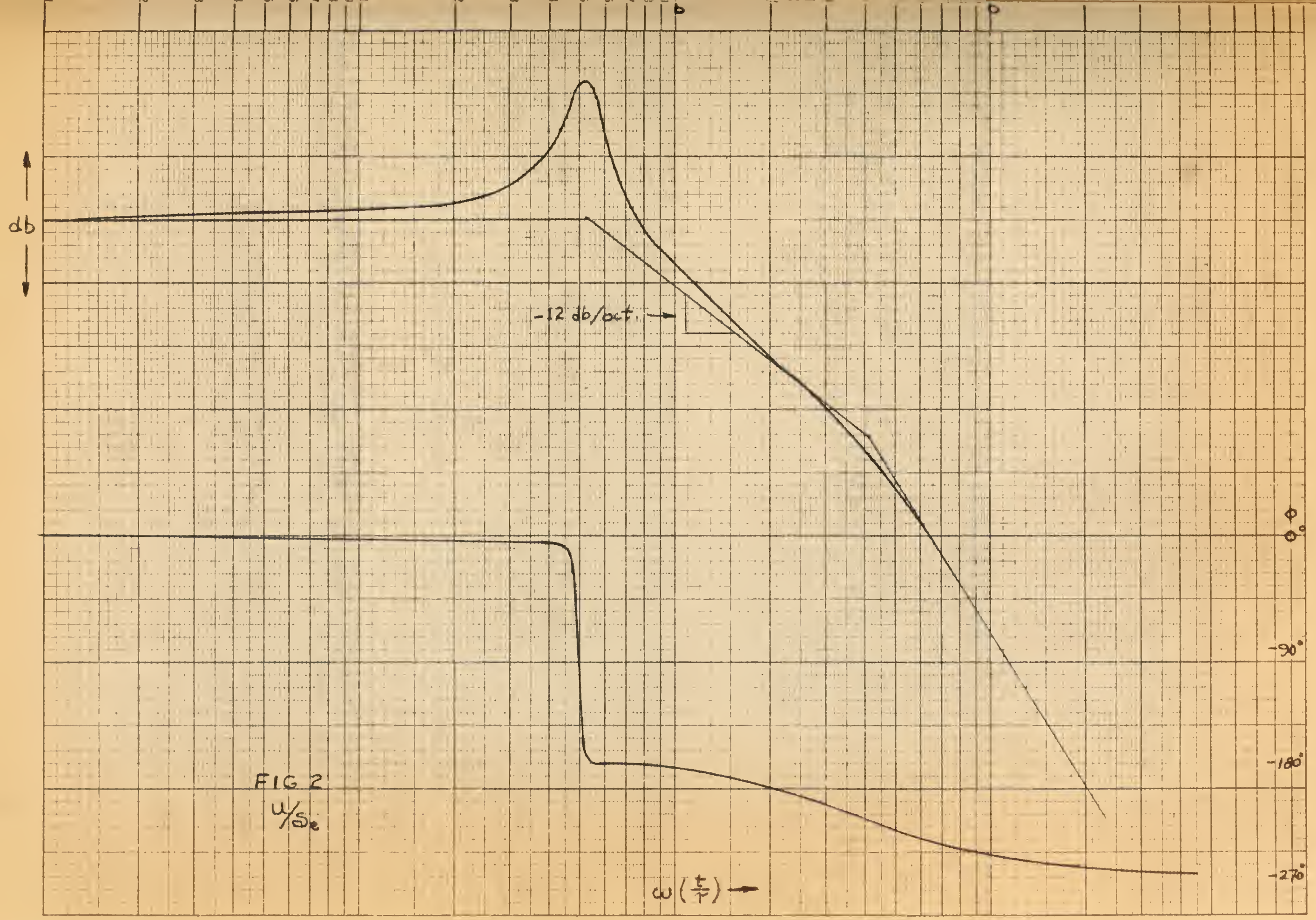


FIG 2

u/δ_e

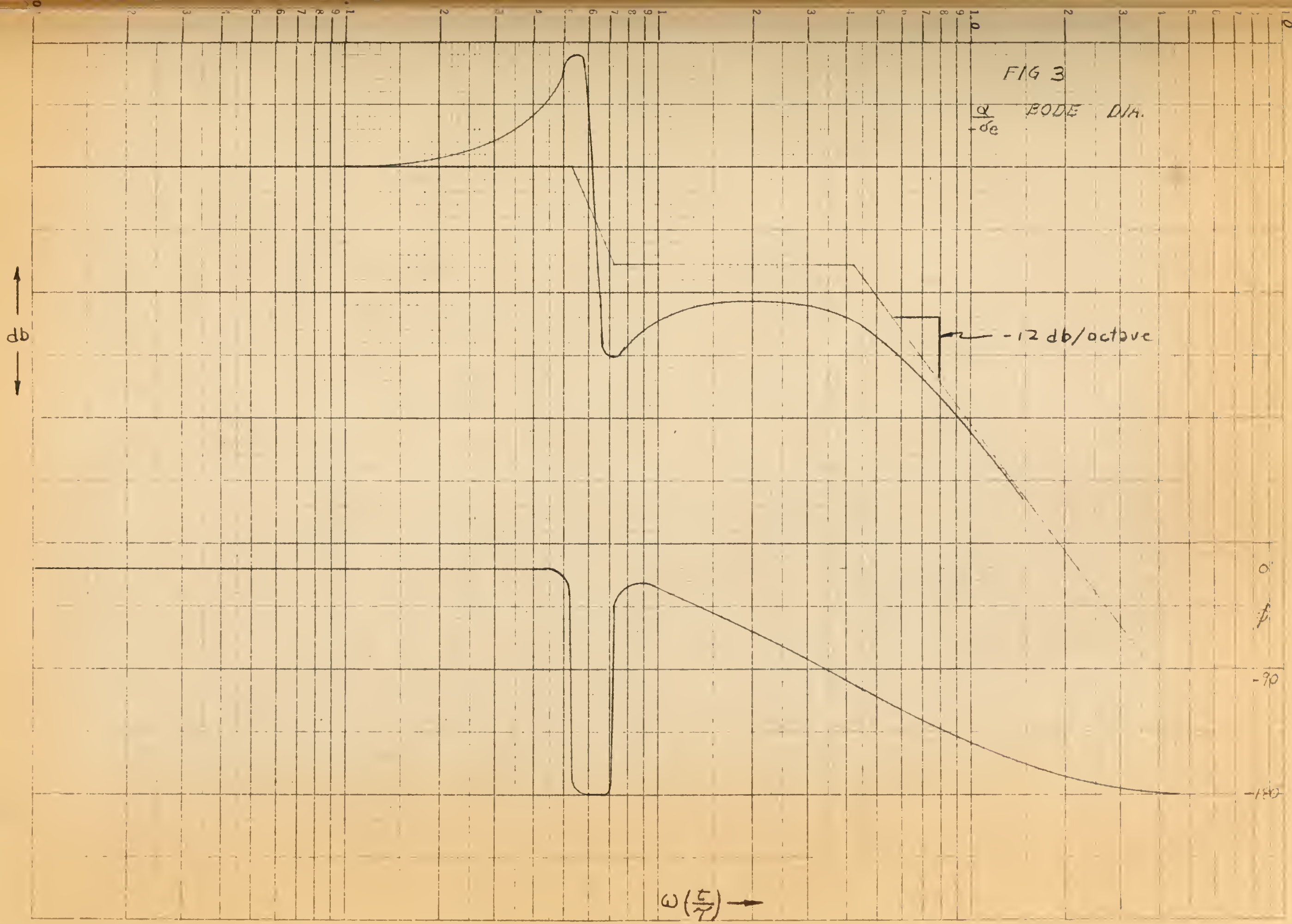


FIG. 4

$\frac{\theta}{-\delta_e}$ BODE DIA

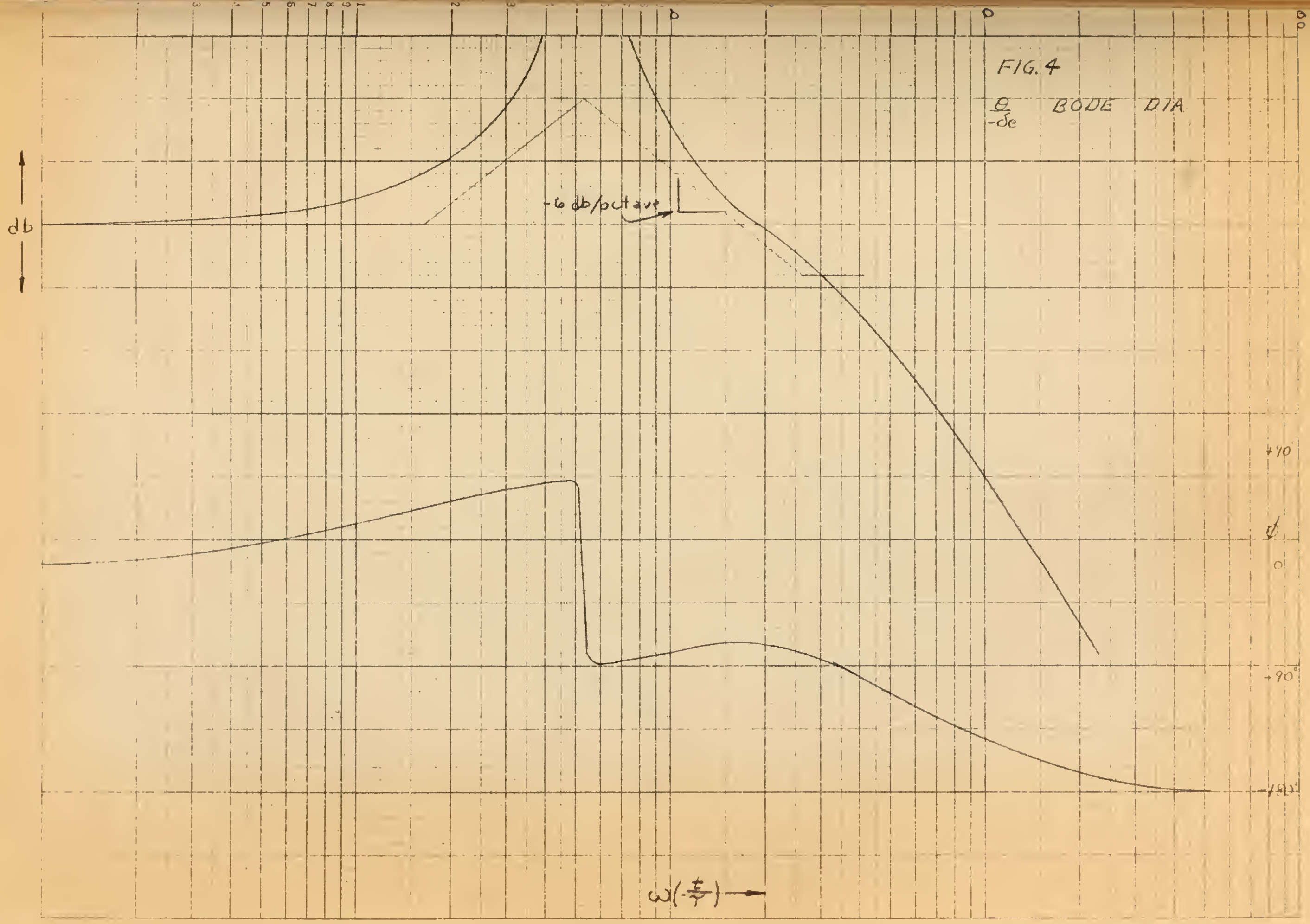
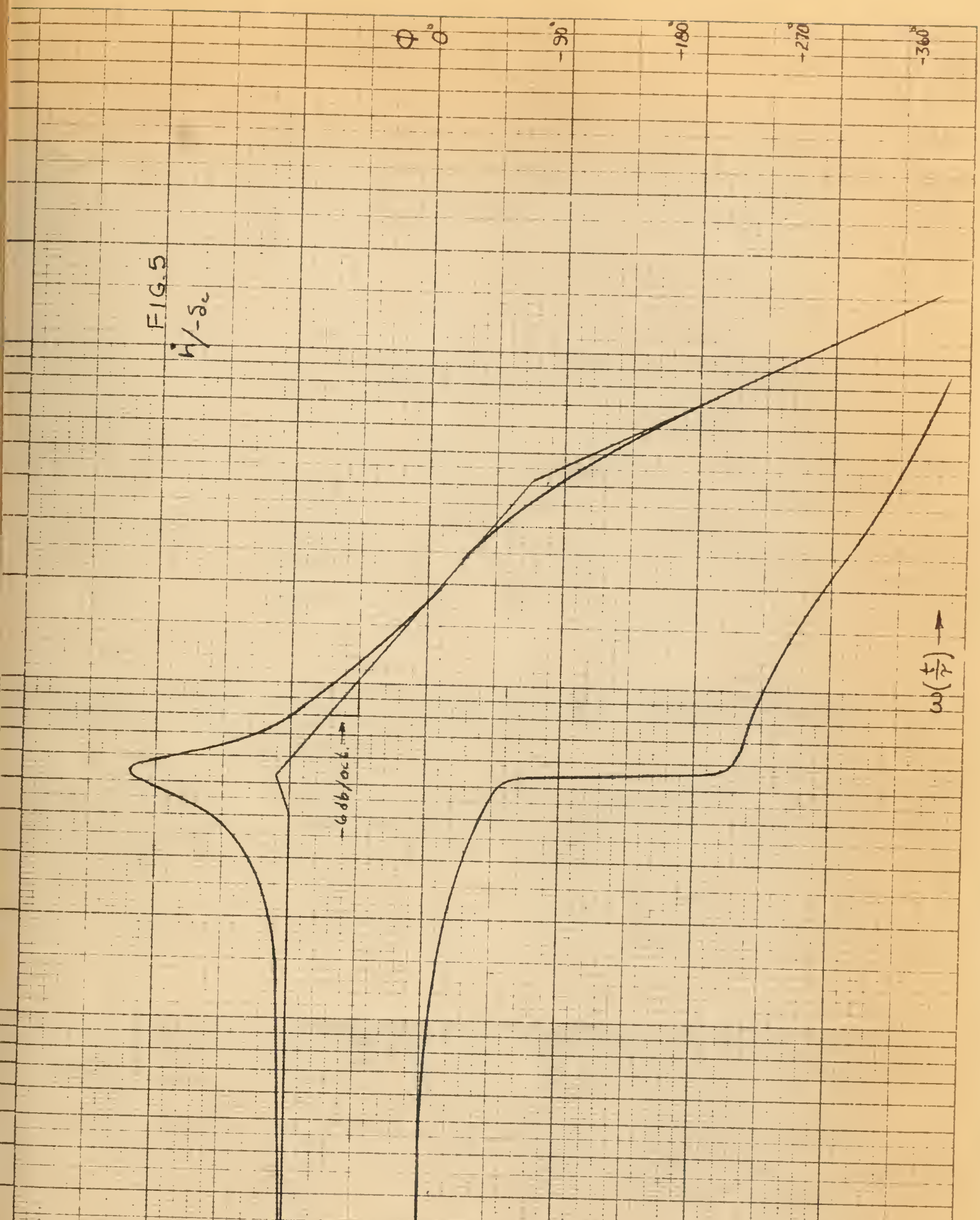


FIG. 5


$\dot{h}/-\delta_c$
Φ
0°
-90°
-180°
-270°
-360°
-6db/oct.
$\omega\left(\frac{t}{\tau}\right)$ →
db

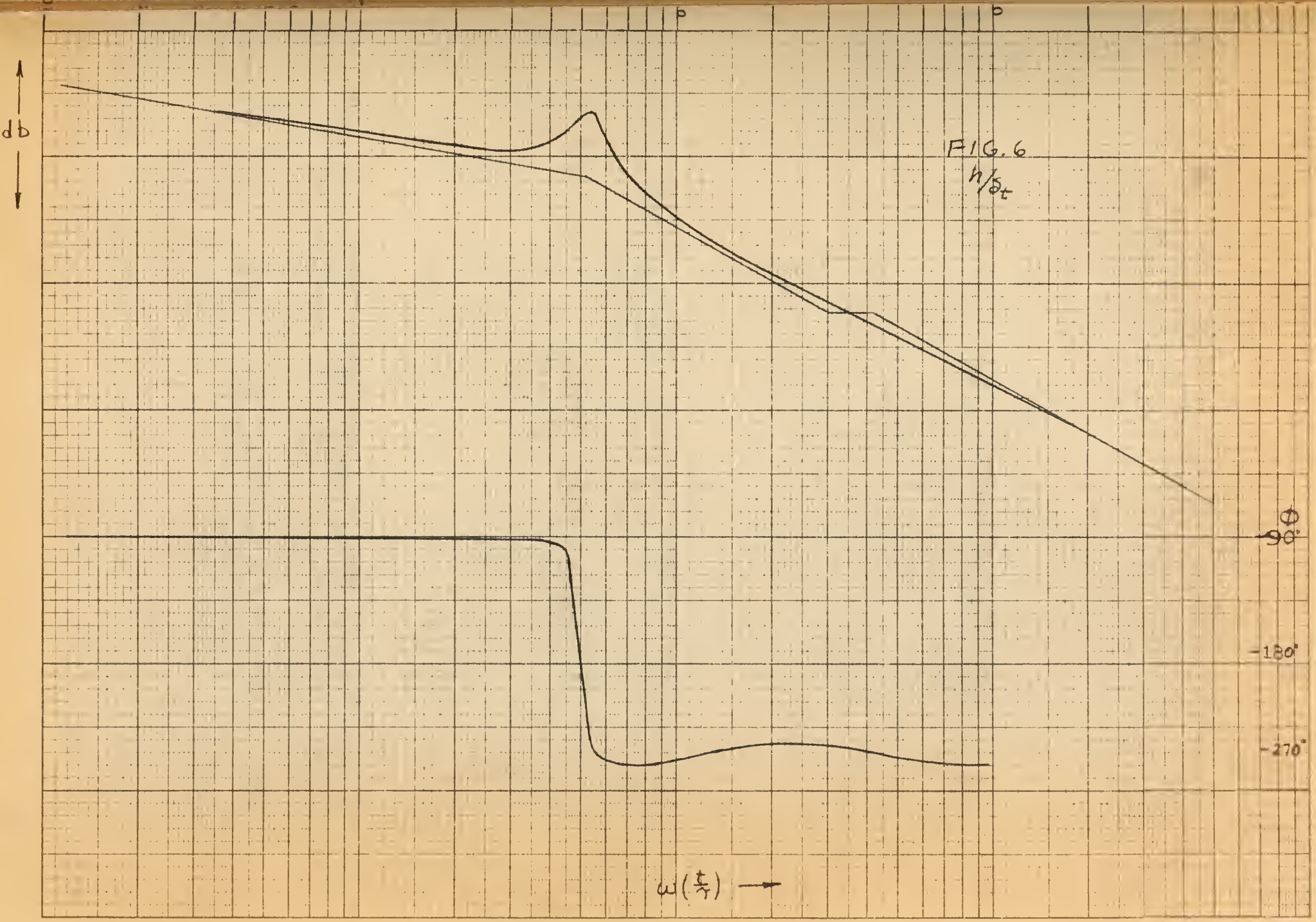


FIG. 6

h/δ_t

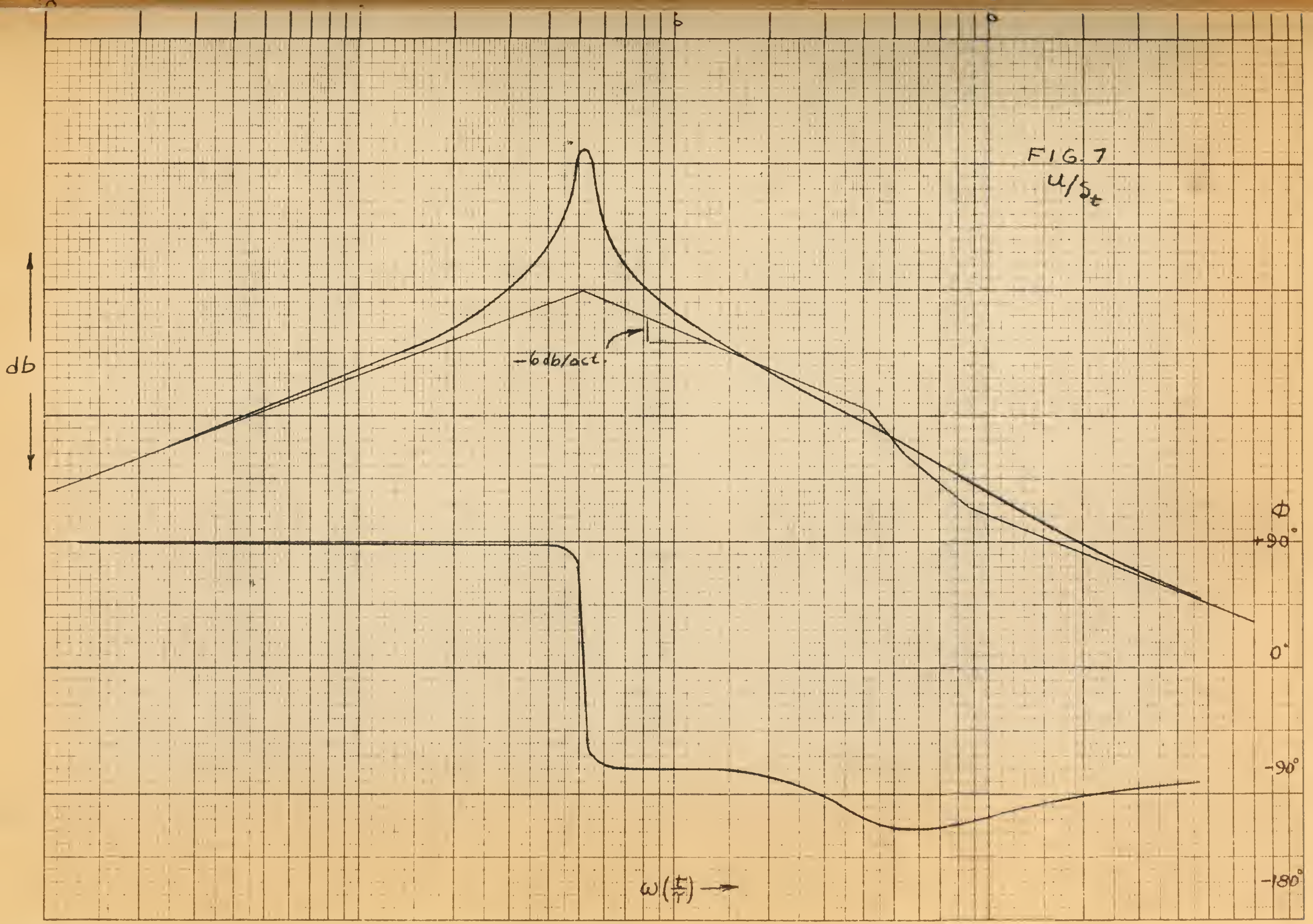
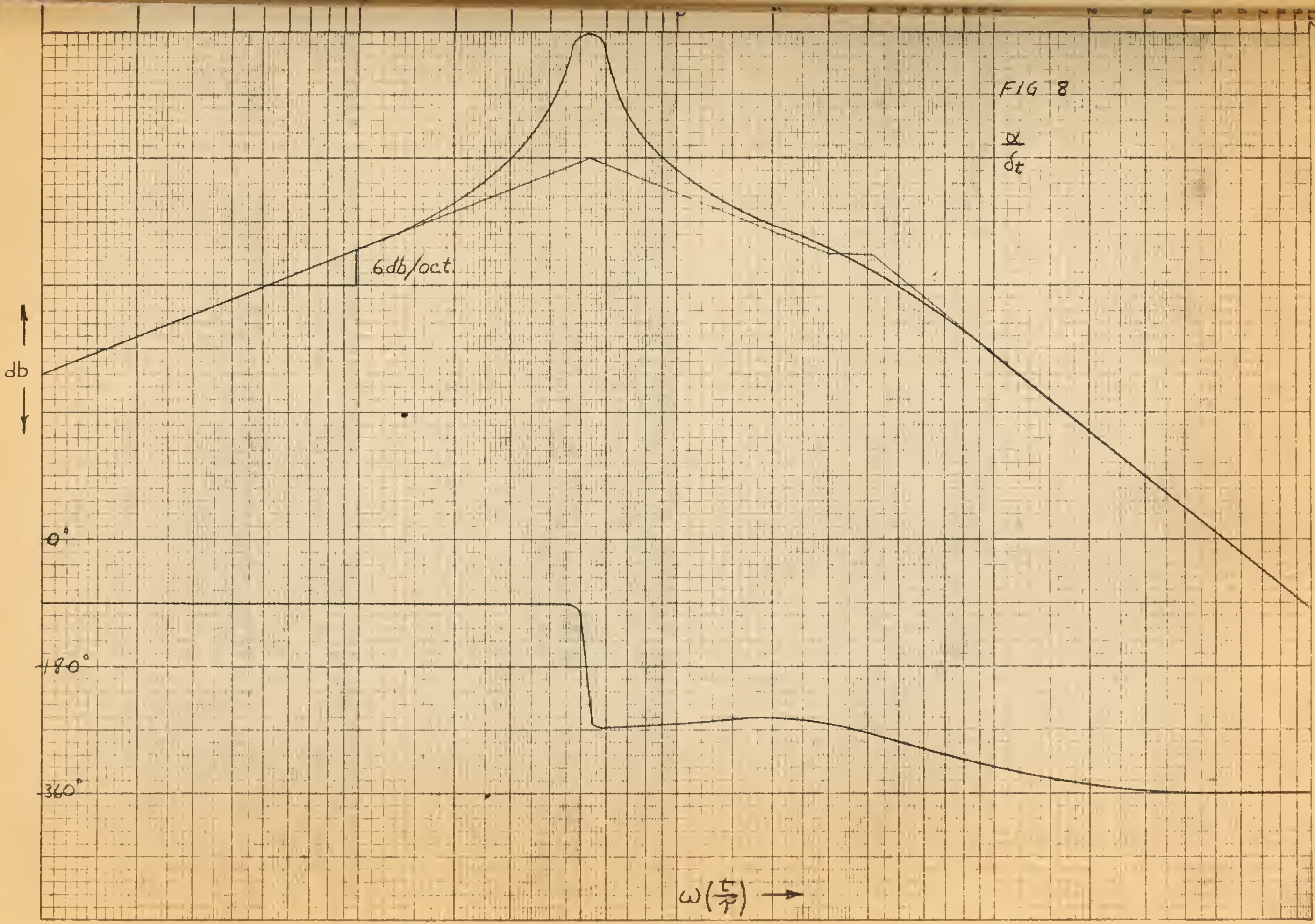


FIG. 7
u/δ_t

FIG 8


$\frac{\alpha}{\delta_t}$
6db/oct.
db
0°
180°
360°
$\omega\left(\frac{t}{\tau}\right) \rightarrow$

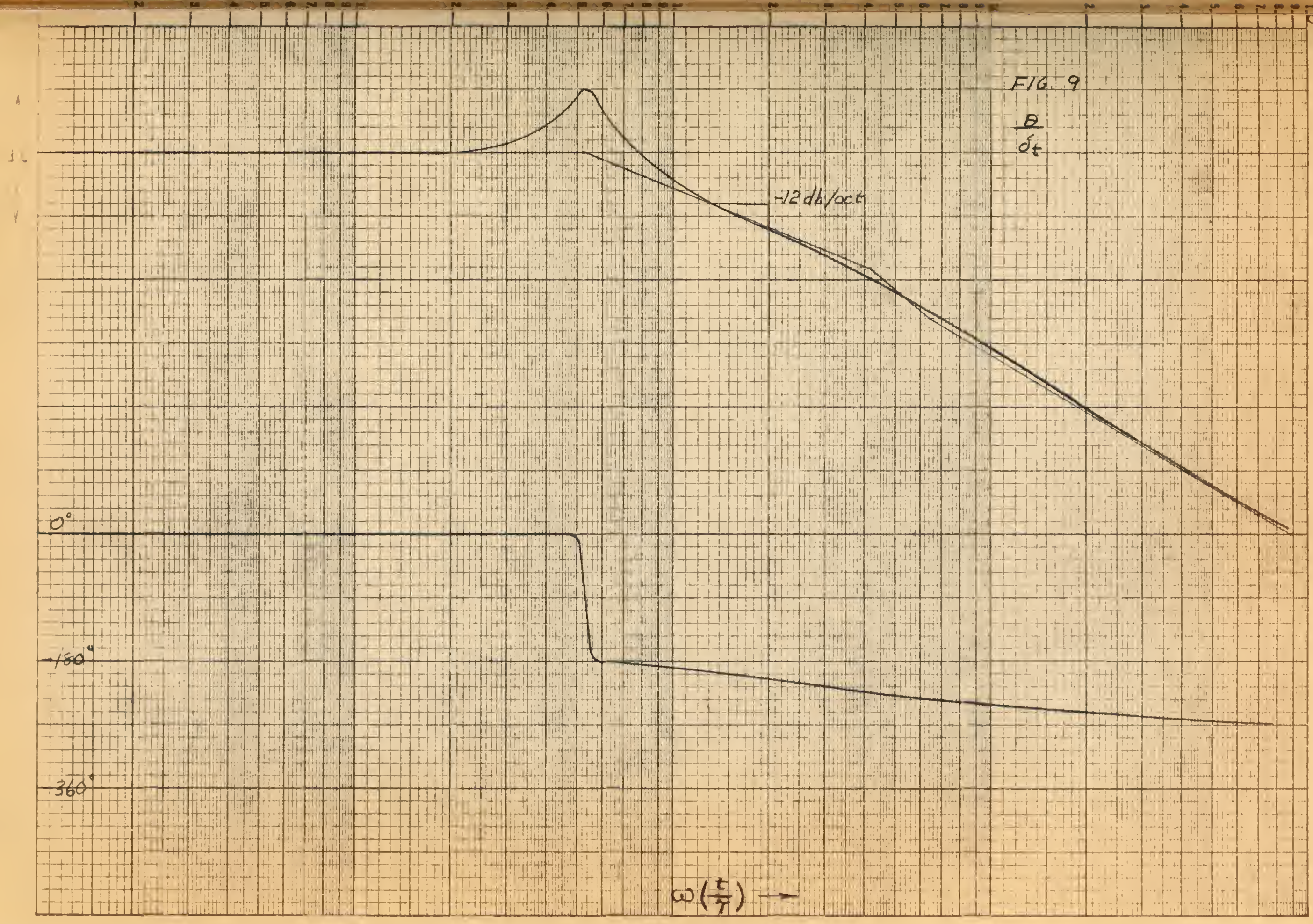


FIG. 9

$\frac{\theta}{\delta_t}$
-12 db/oct
0°
-180°
-360°
$\omega\left(\frac{t}{T}\right) \rightarrow$

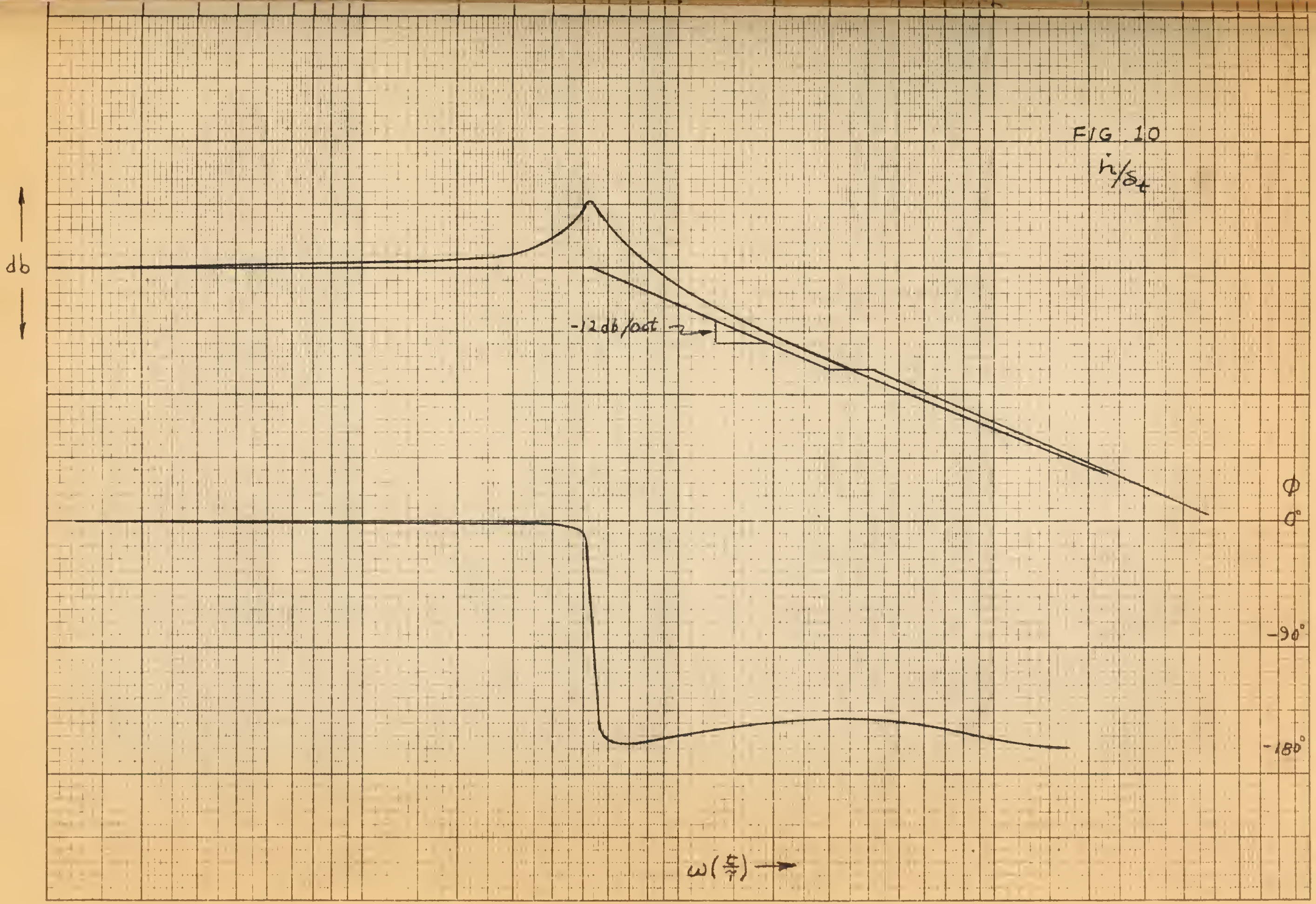


FIG 10

$\dot{h}/\delta_t$

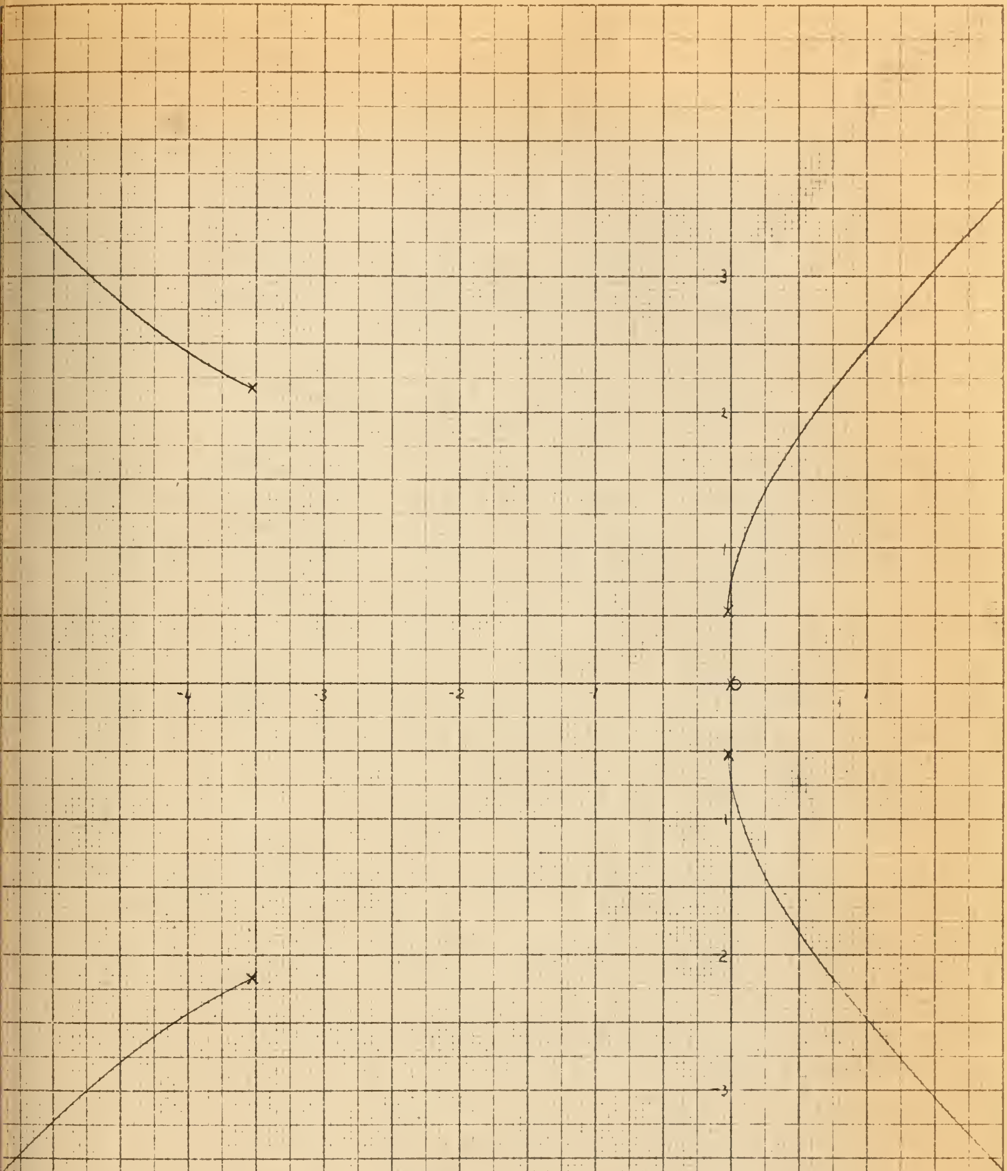


FIG 11

$h/-S_e$ with unity feedback

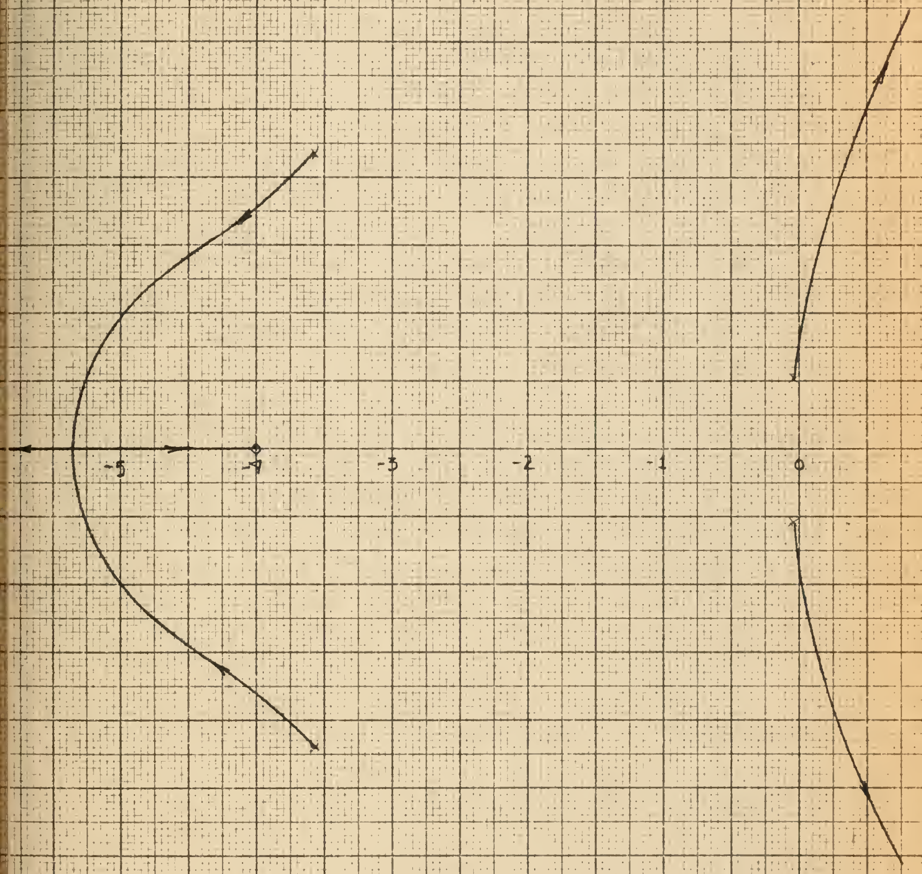
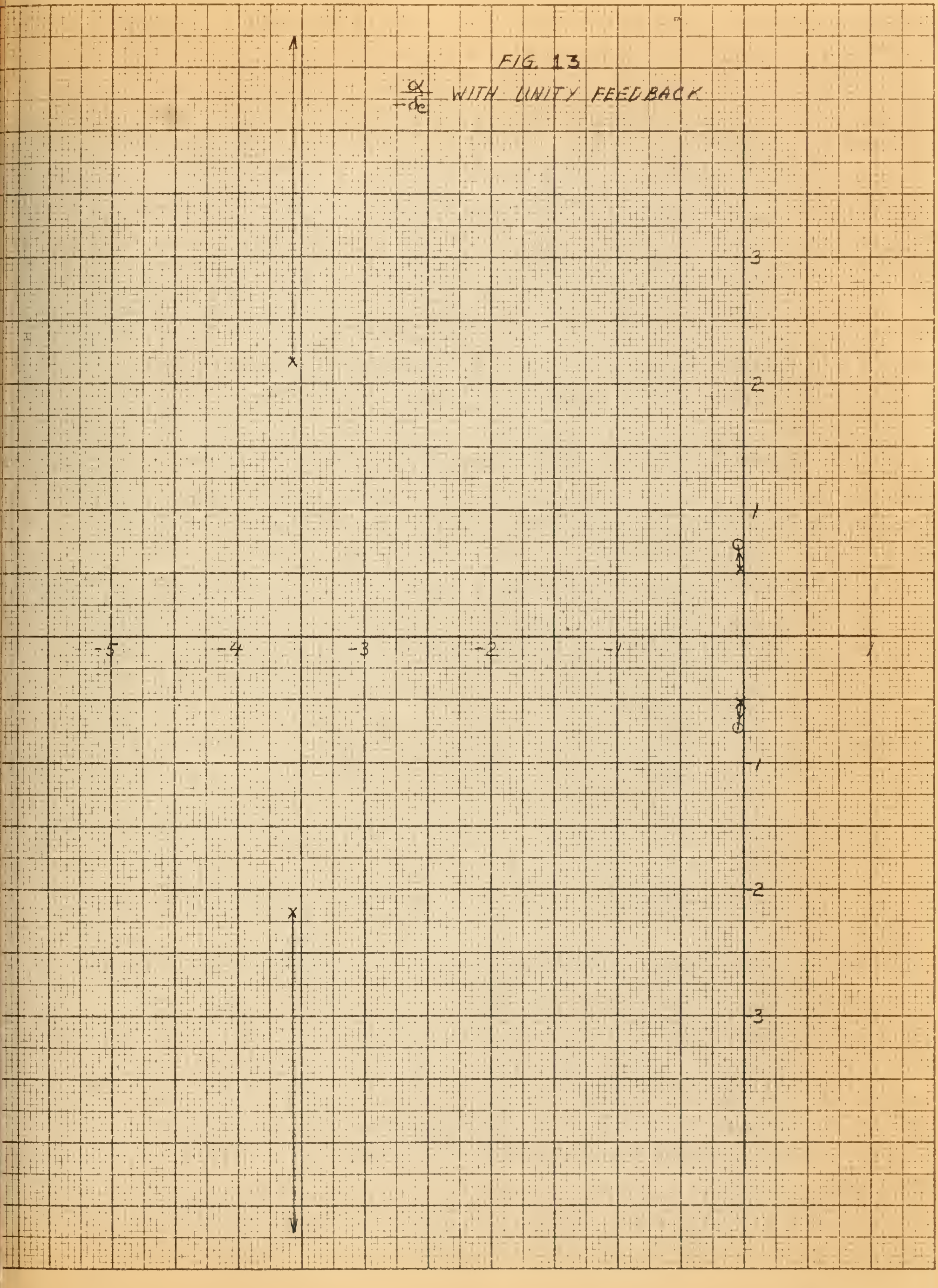


FIG 12

u/δ_e with unity feedback

FIG. 13

$\frac{\alpha}{-\delta_e}$ WITH UNITY FEEDBACK



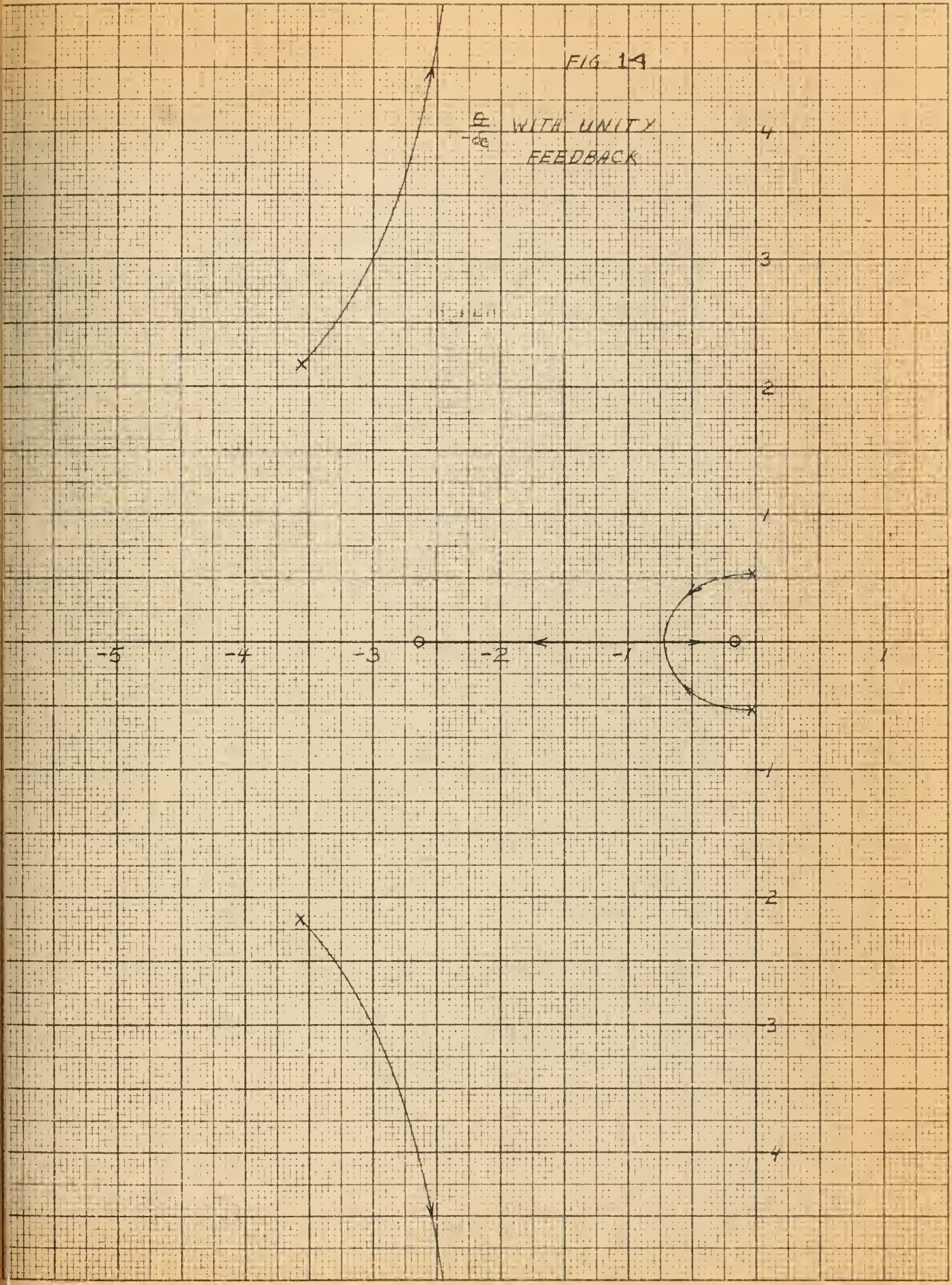


FIG 14

$\frac{\theta}{-\delta_e}$ WITH UNITY FEEDBACK

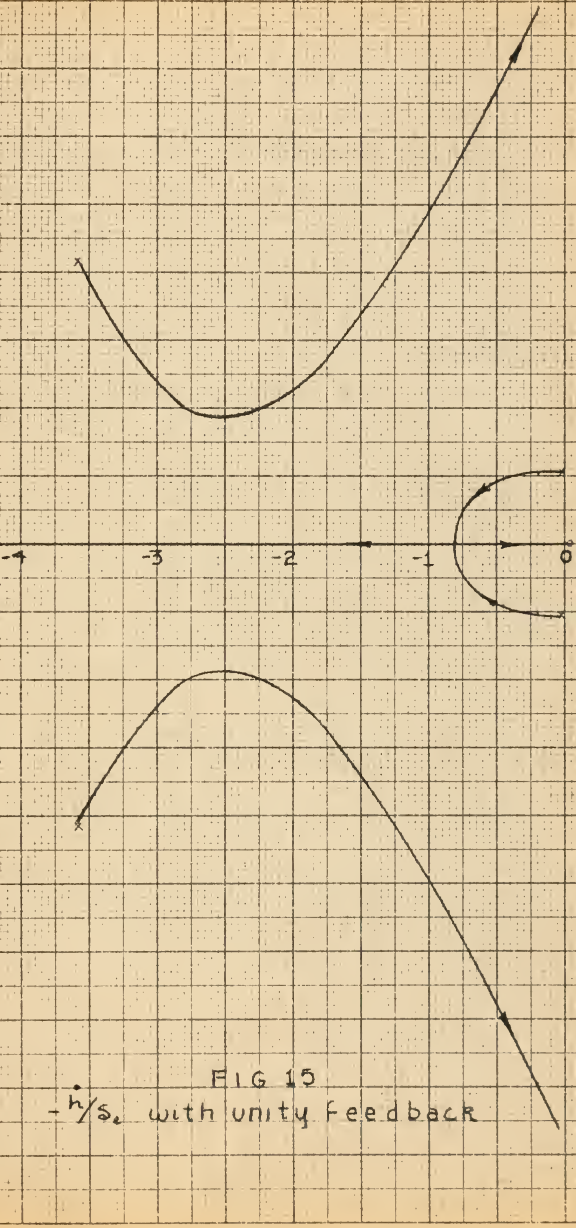


FIG 15

$-\dot{h}/s_e$ with unity feedback

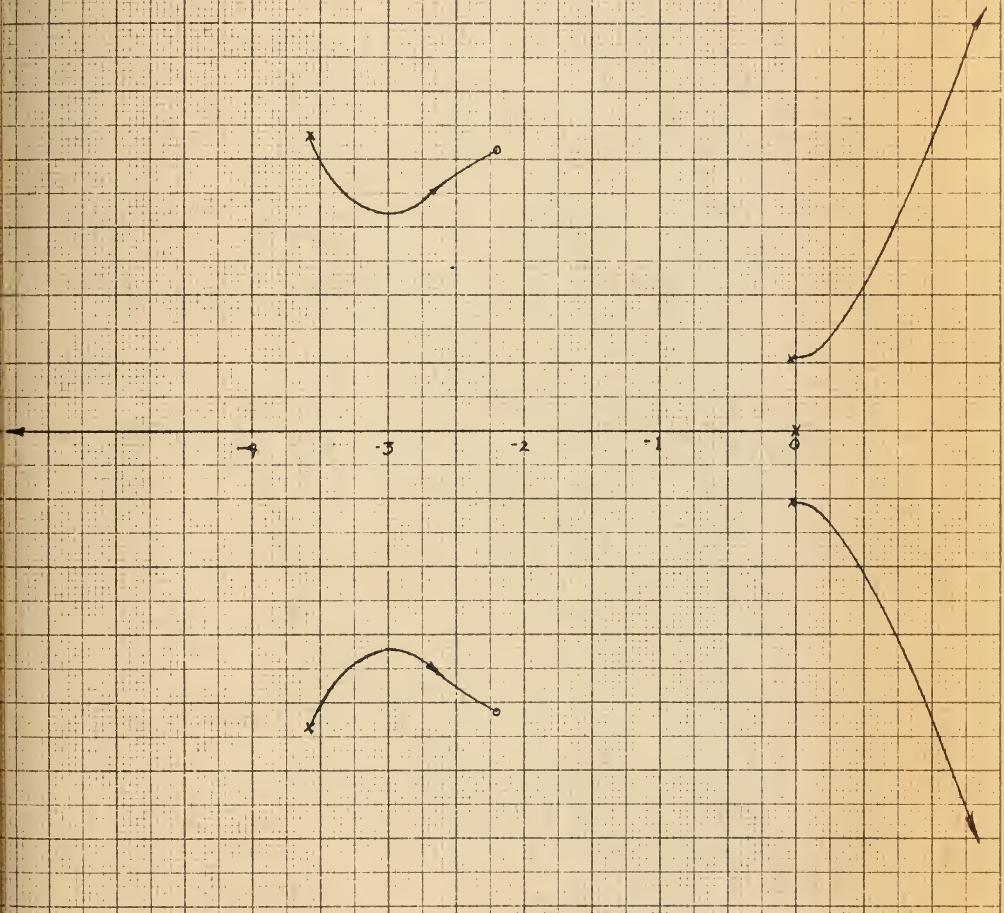


FIG 16

h/S_t with unity feedback

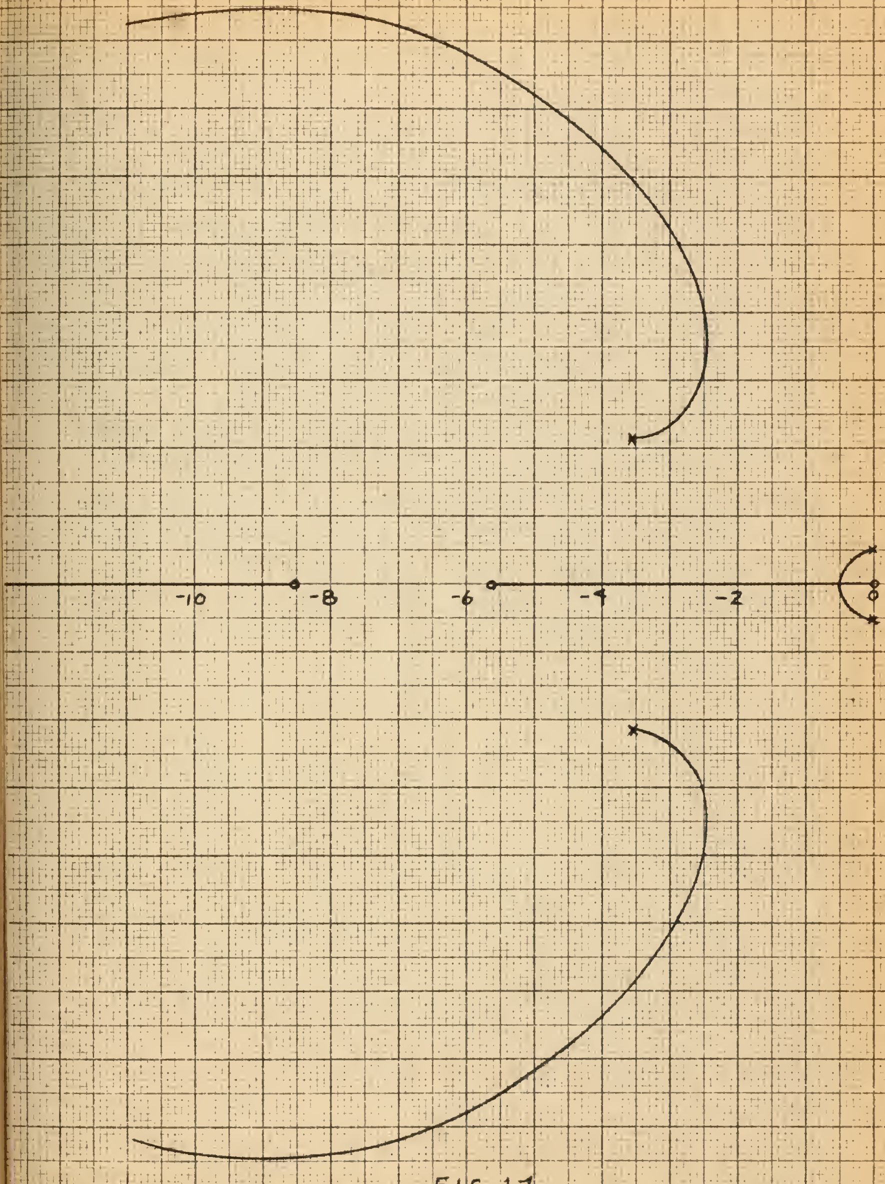


FIG 17

u/δ_t with unity feedback

FIG. 18

$\frac{\alpha}{\delta_t}$ WITH UNITY FEEDBACK

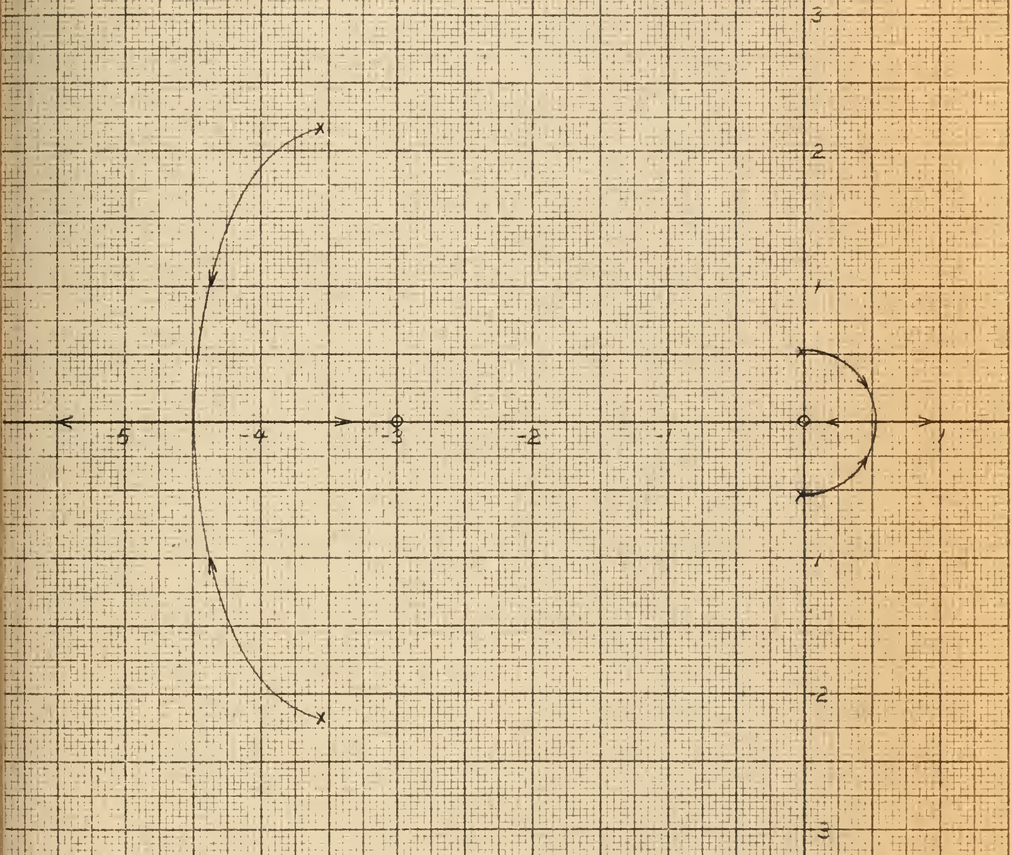
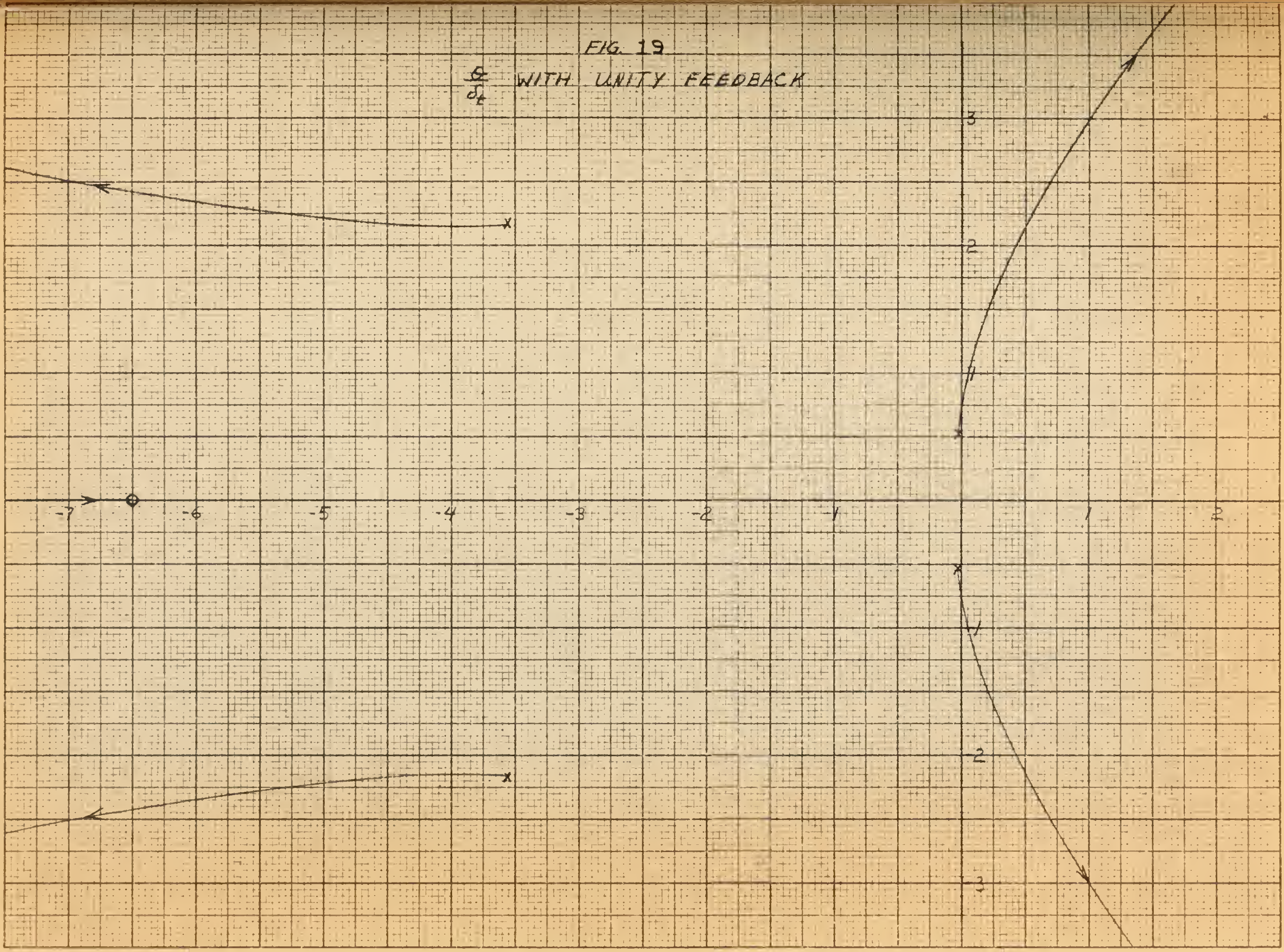


FIG. 19

$\frac{\theta}{\delta_t}$ WITH UNITY FEEDBACK



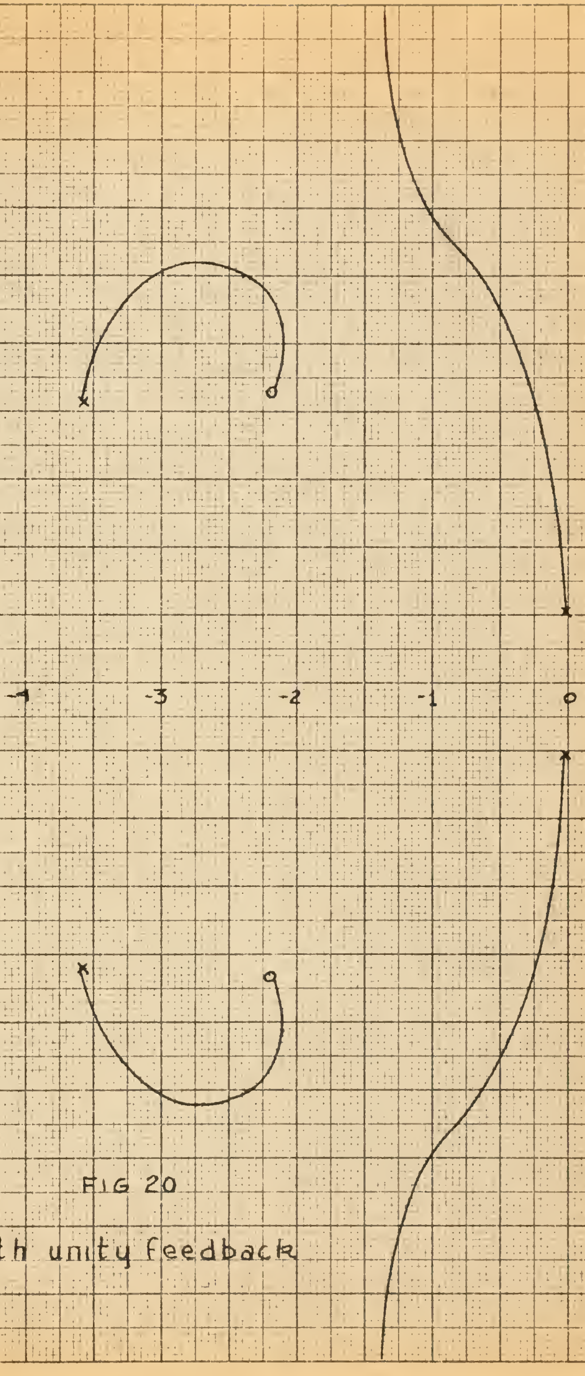
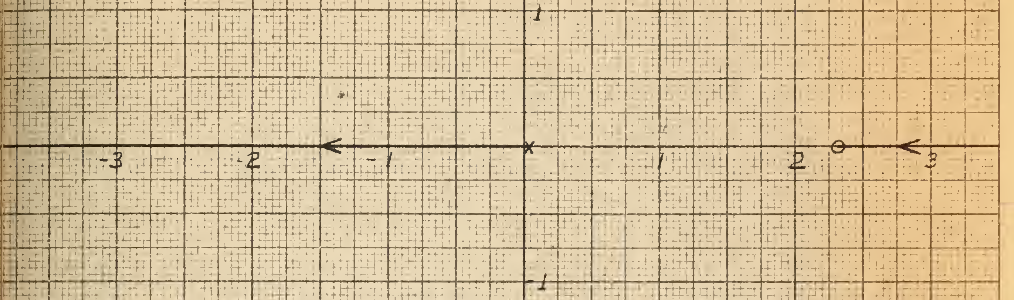


FIG 20

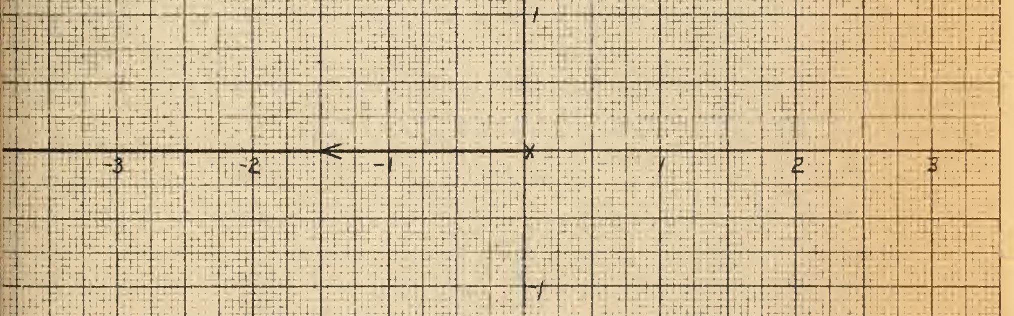
h/S_t with unity feedback

FIG. 21

ROOT LOCI OF NUMERATOR OF h/δ_e
FOR SEVERAL FEEDBACKS



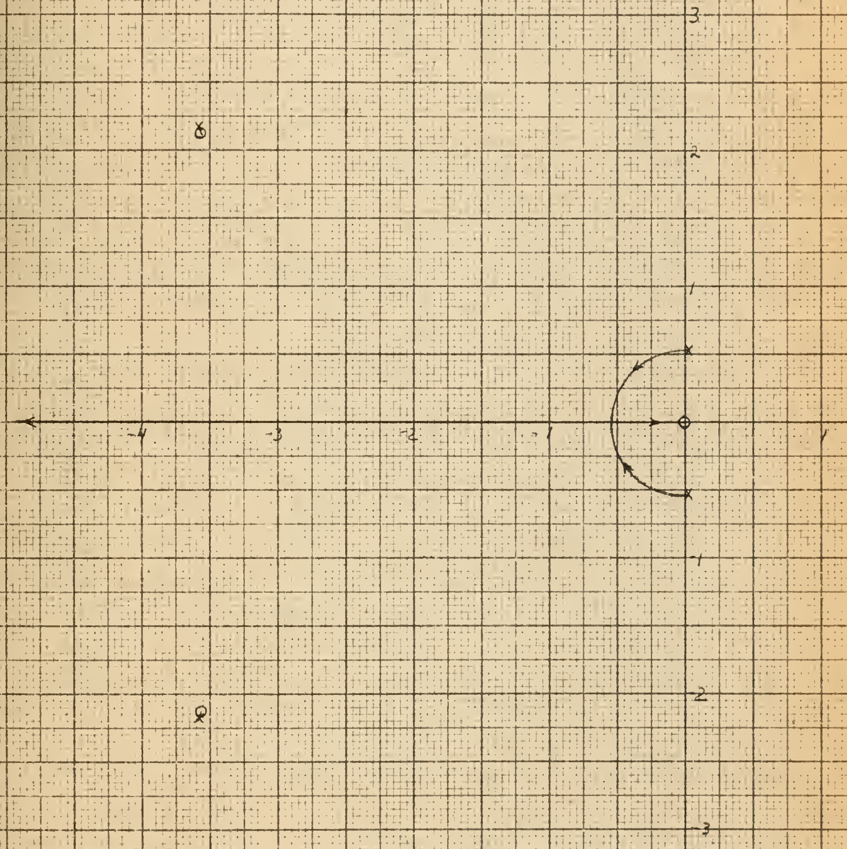
ELEVATOR POSITION FEEDBACK



ANGLE OF ATTACK AND VELOCITY FEEDBACK

FIG. 22

ROOTS OF CHARACTERISTIC
EQUATION FOR $C_b \nearrow$



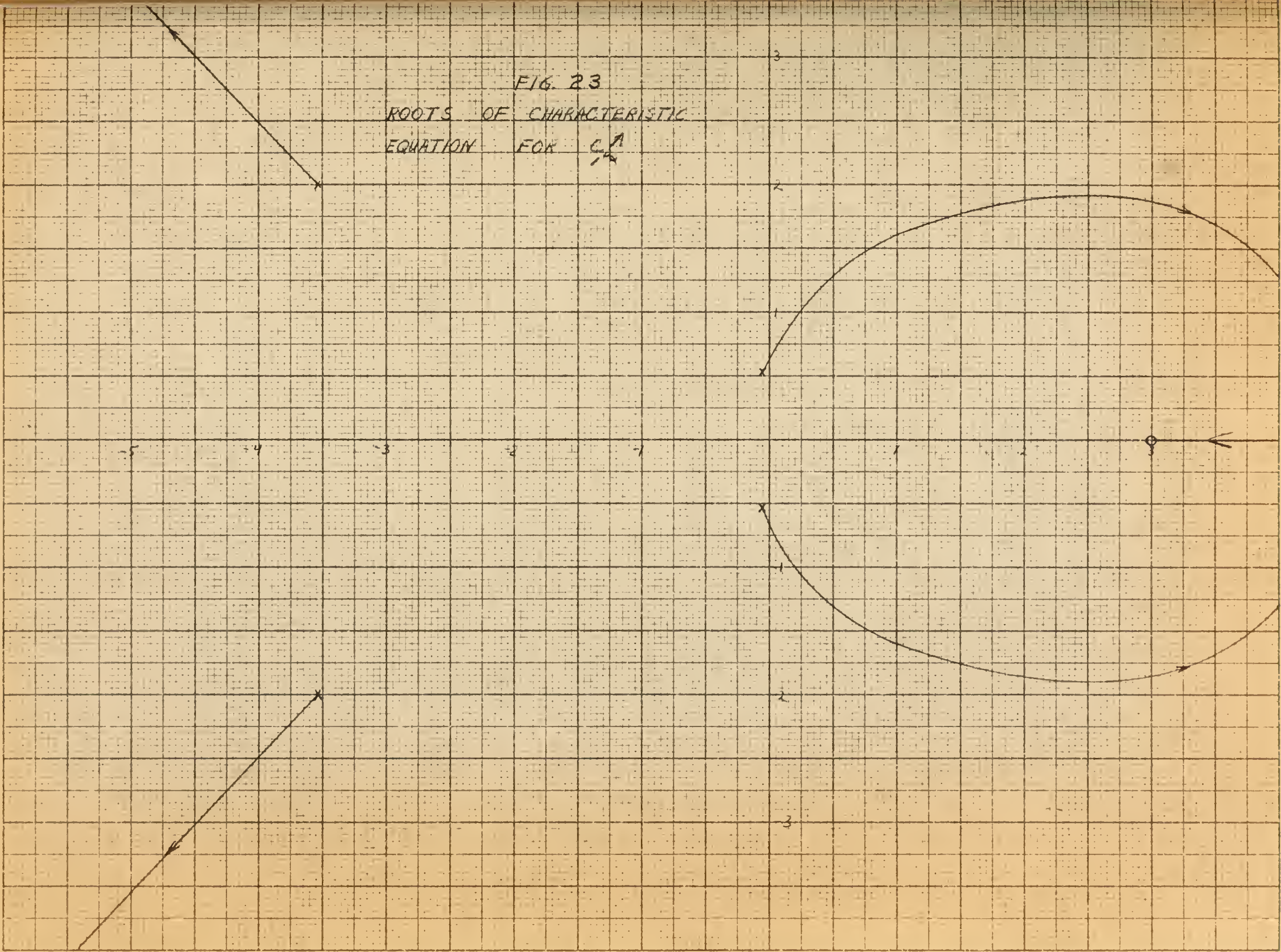


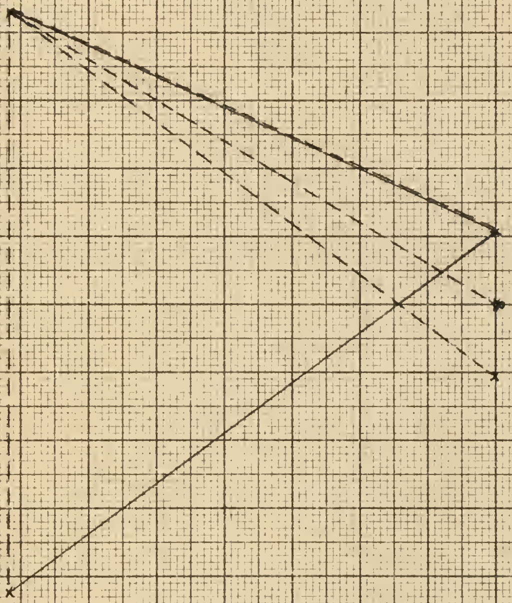
FIG. 23

ROOTS OF CHARACTERISTIC EQUATION FOR C_{ℓ_A}

FIG. 24

TRANSIENT RESPONSE EVALUATION

$C_D = .092$

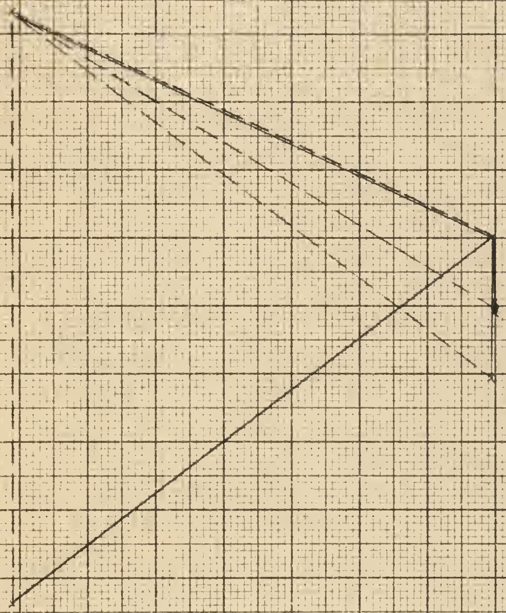


$$h(t) = -8650\left[+.209 + .0086t + .2016e^{-.024t}\cos(.53t - 191.5^\circ) + .00624e^{-3.58t}\cos(2.17t - 177^\circ)\right]$$

FIG. 25

TRANSIENT RESPONSE EVALUATION

$C_0 = .1345$

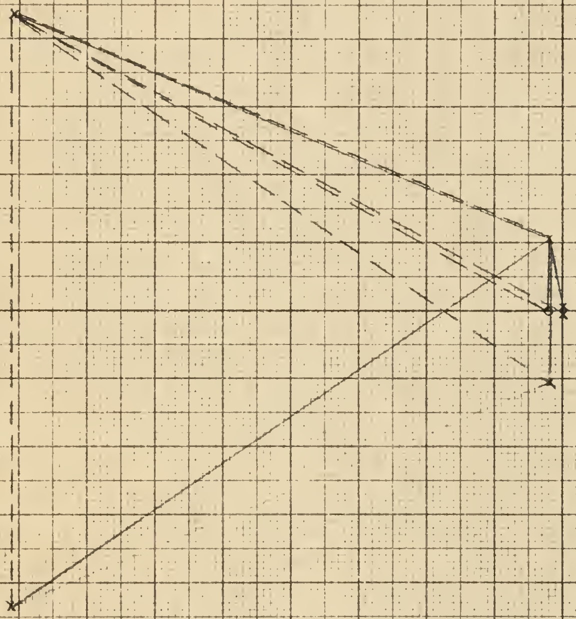


$$h(t) = -8650\left[+.205 + .212\,e^{-.04t}\cos(.53t - 192^\circ) + .0064\,e^{-3.57t}\cos(2.17t - 176^\circ)\right]$$

FIG. 26

TRANSIENT RESPONSE EVALUATION

$C_D = .290$



$$h(t) = -8650\left[.1735 + .0207t + .202e^{-.1t}\cos(.525t - 213^\circ) + .0063e^{-3.57t}\cos(2.17t - 177^\circ)\right]$$

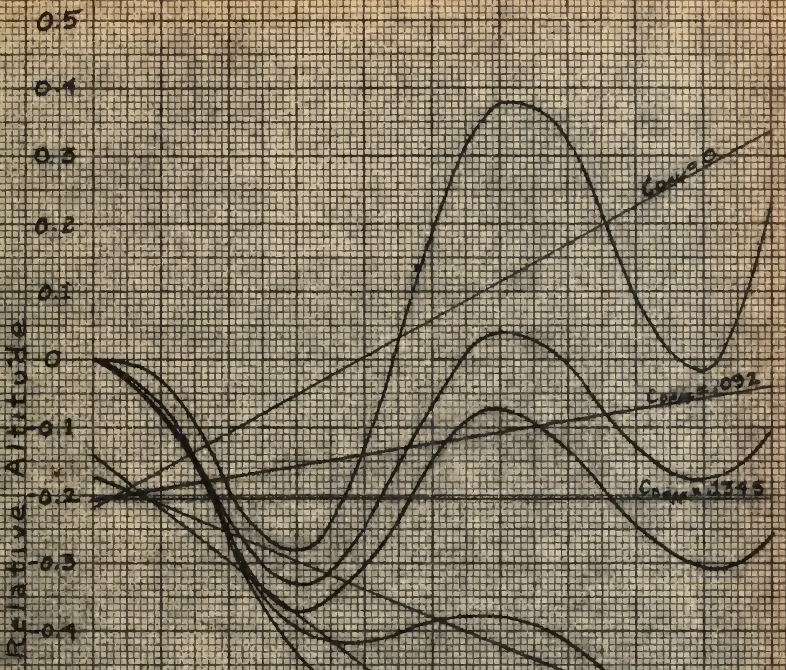
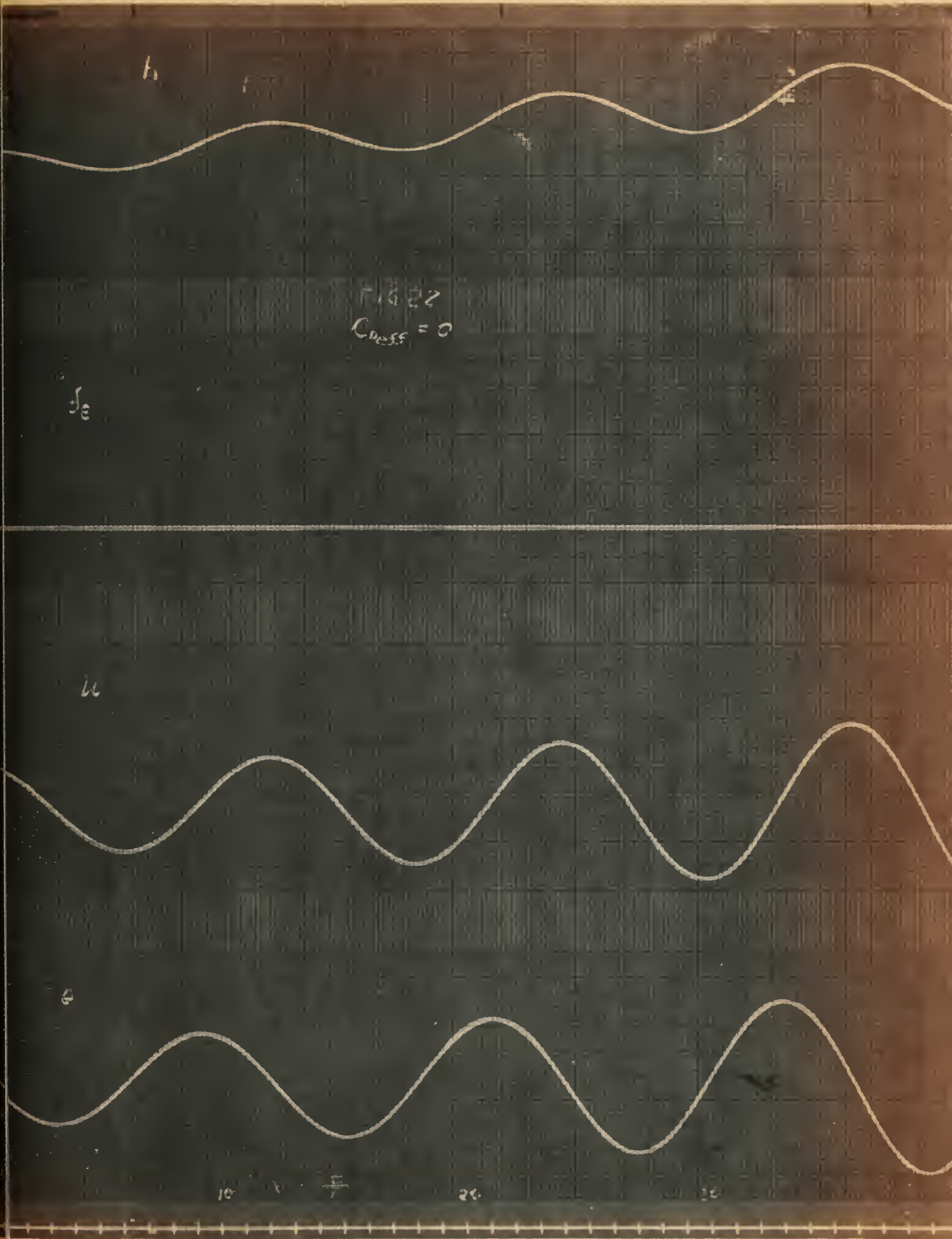


FIG 27
PLOT
of
RELATIVE ALTITUDE VARIATION
versus
NON DIMENSIONAL TIME
at
$C_{D_{off}}$ = CONSTANT


h
FIG 82
$C_{\text{press}} = 0$
δ_E
u
θ
10
20
30

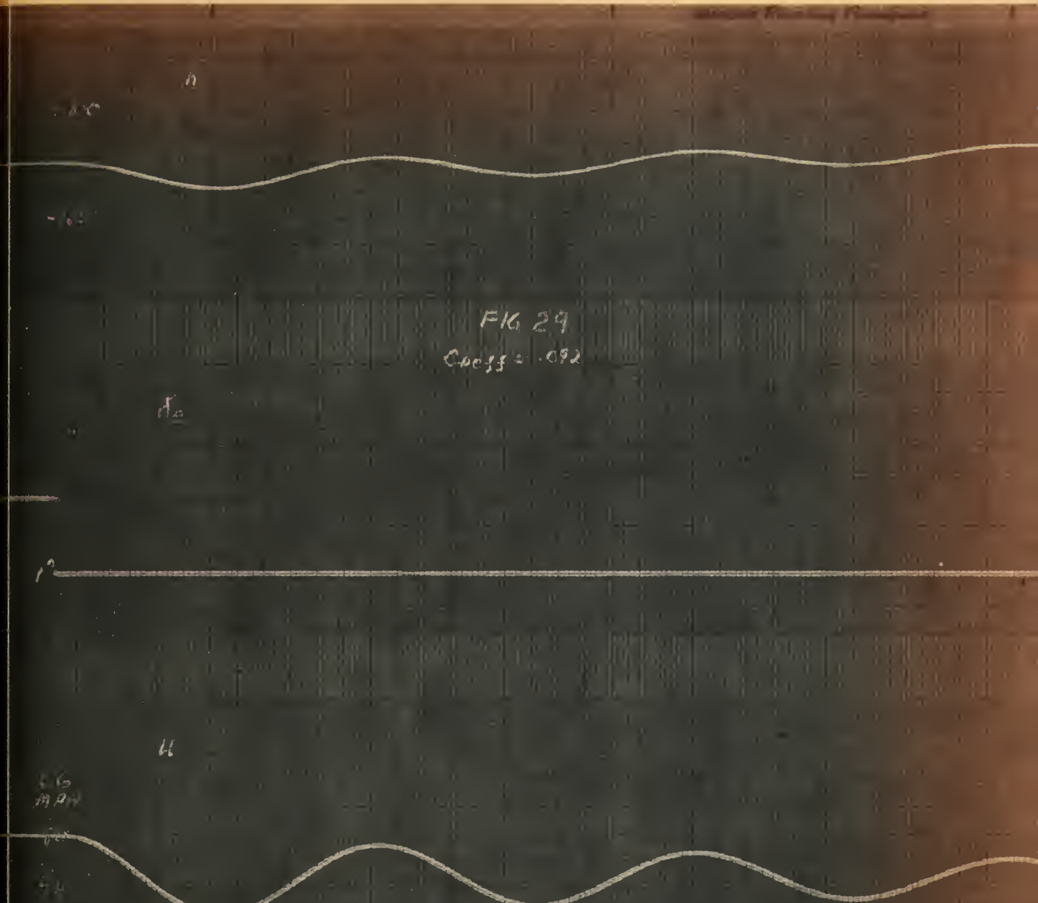


FIG. 29
Coeff = .092

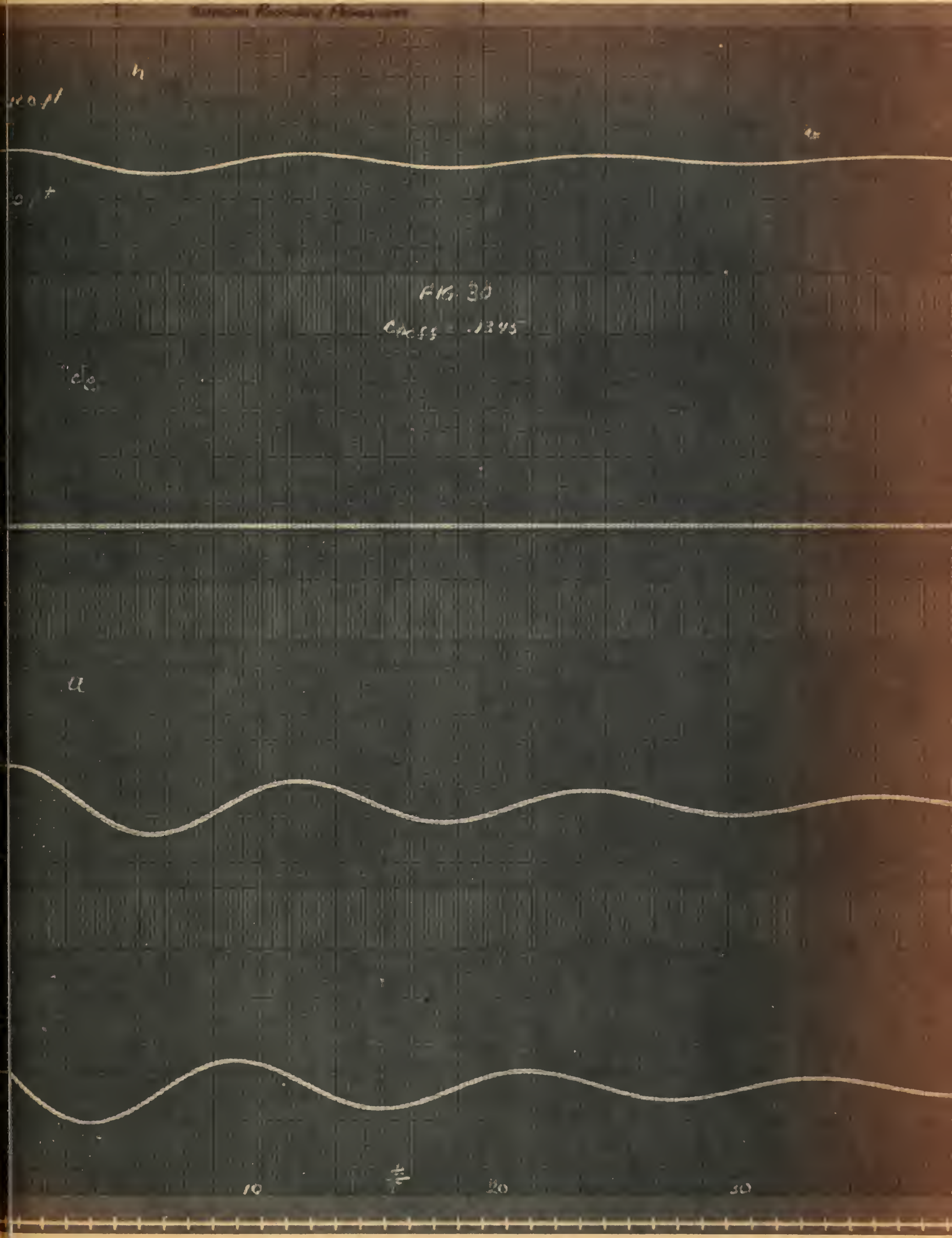
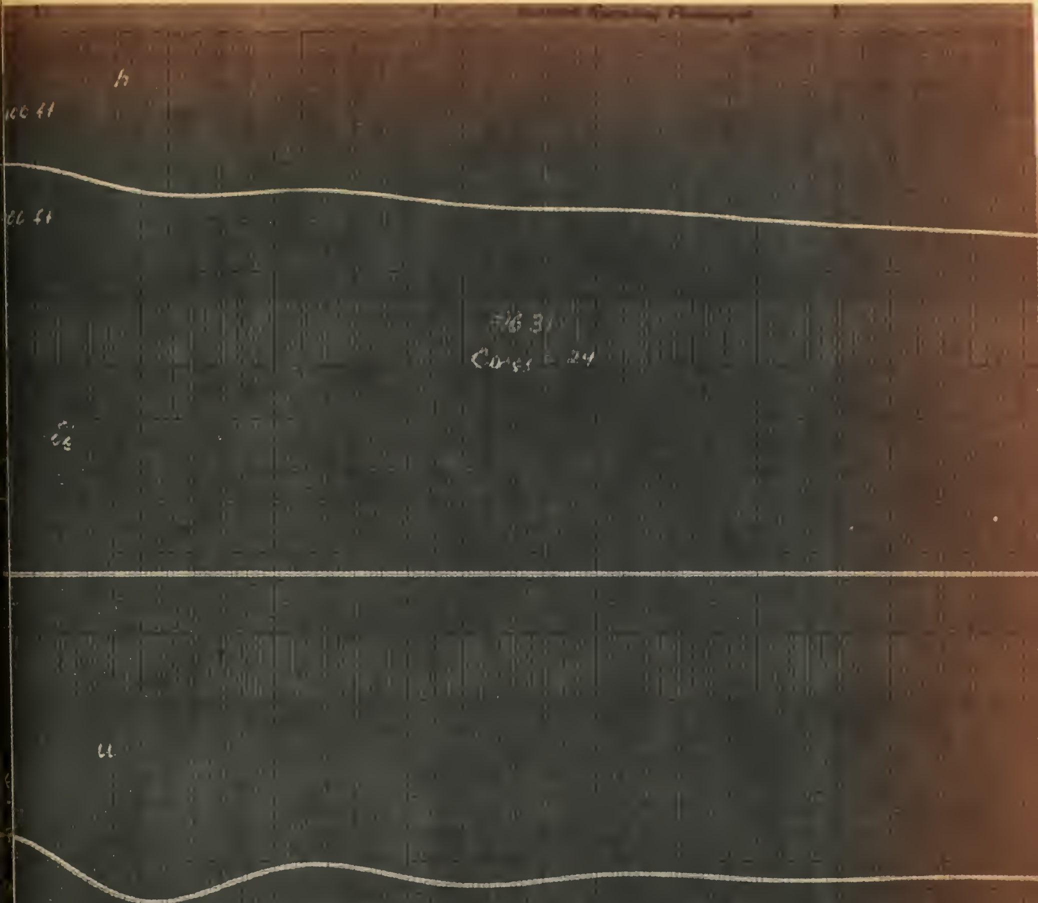


FIG. 30

Cmass = .1345


h
00 ft
00 ft
FIG 31
Case 24
i_g
u
θ
10
20
30

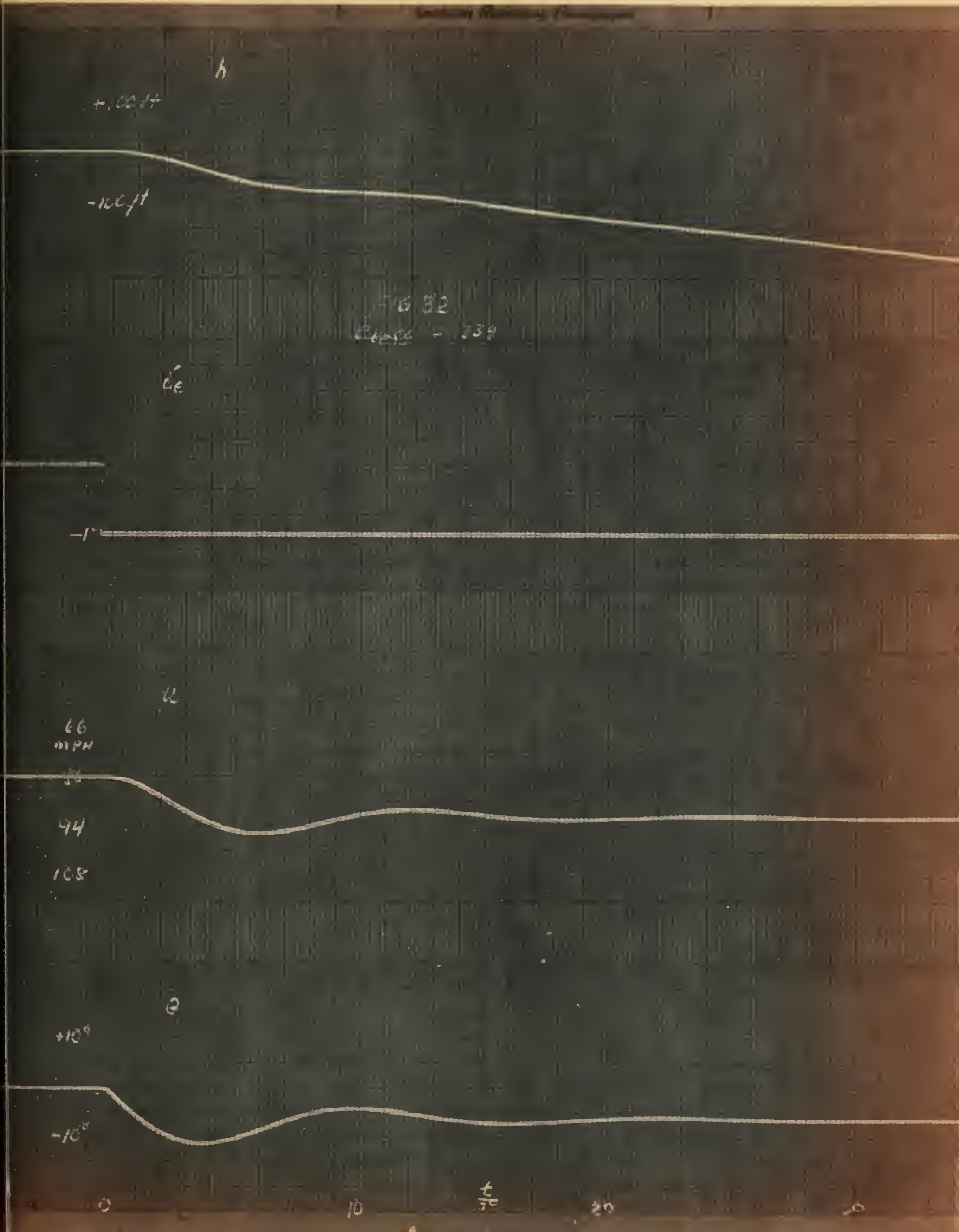


FIG 32

[illegible] - .734

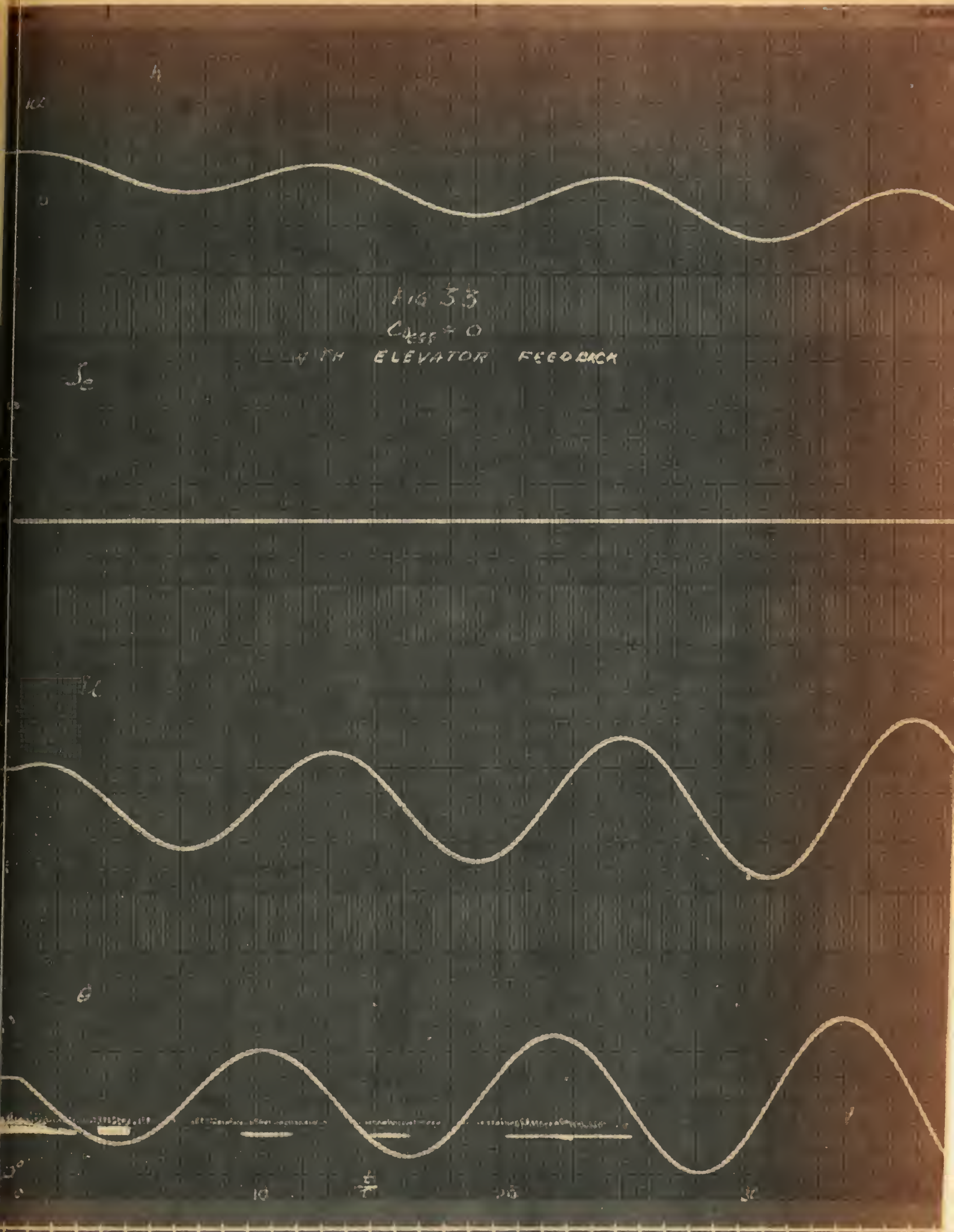


FIG 35
$C_{h_{eff}} = 0$
WITH ELEVATOR FEEDBACK

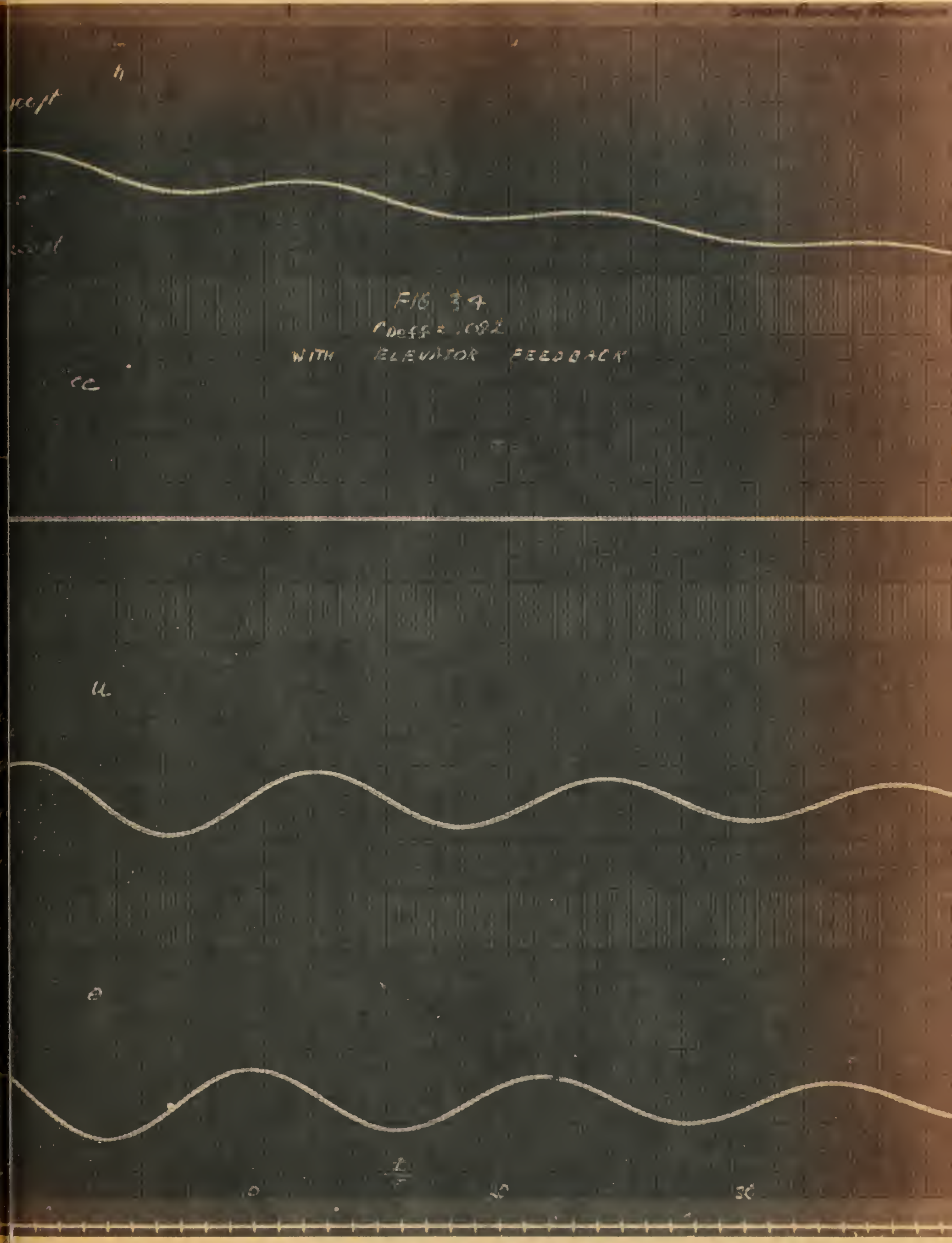


FIG 37
$r_{Deff} = .082$
WITH ELEVATOR FEEDBACK

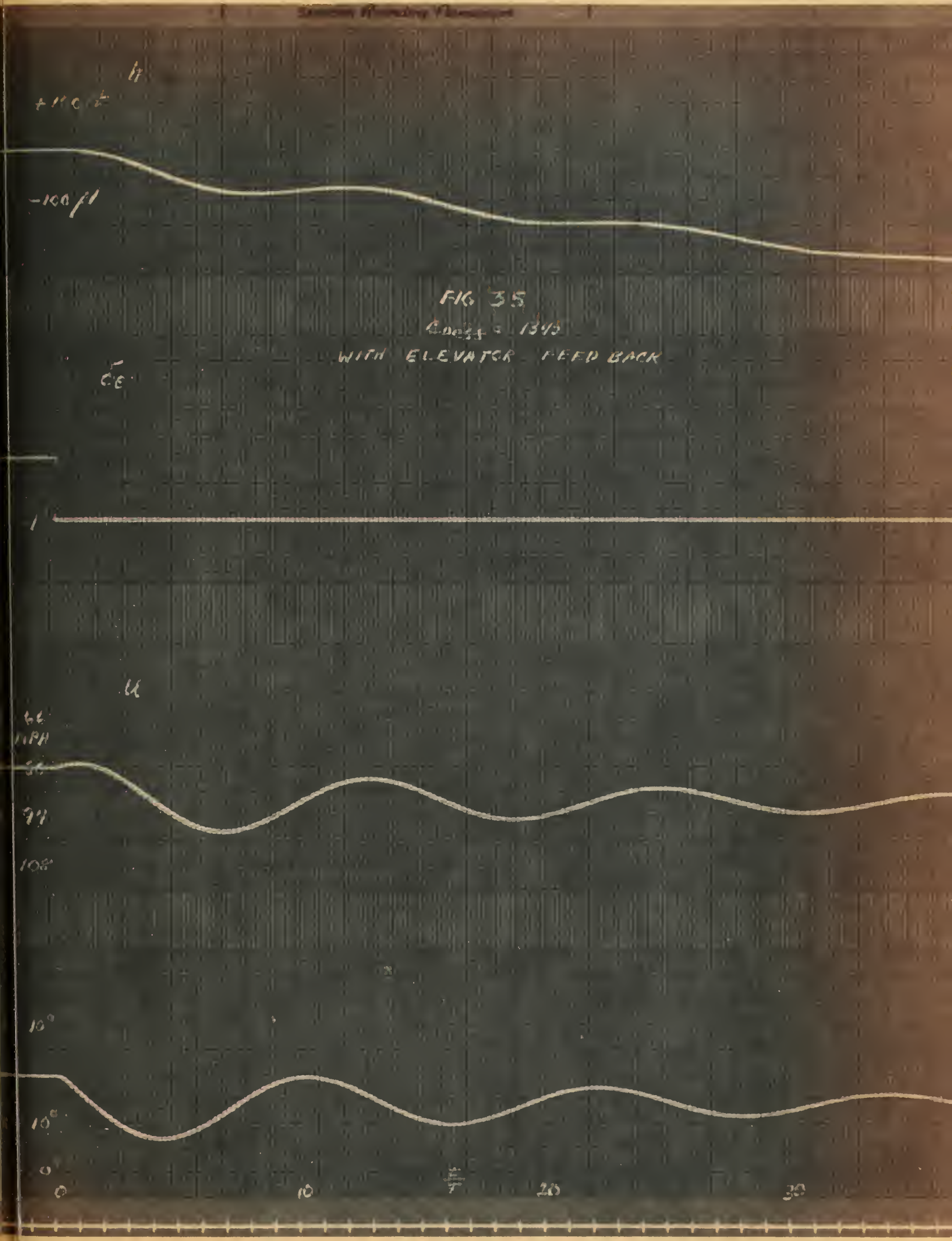


FIG 35

[illegible] = 1343

WITH ELEVATOR FEED BACK

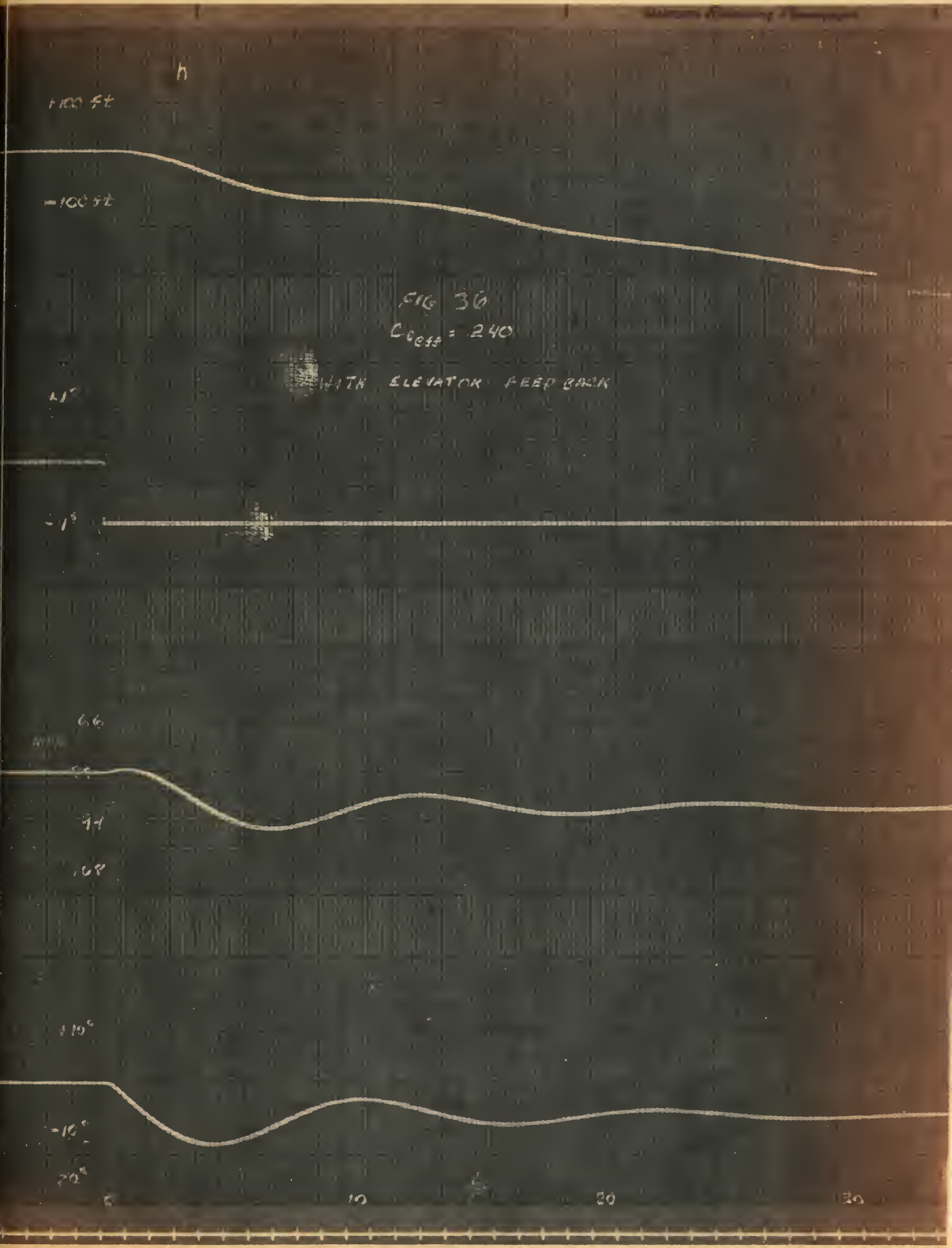


FIG 36

$C_{\ell_{eff}} = .240$

WITH ELEVATOR FEED BACK

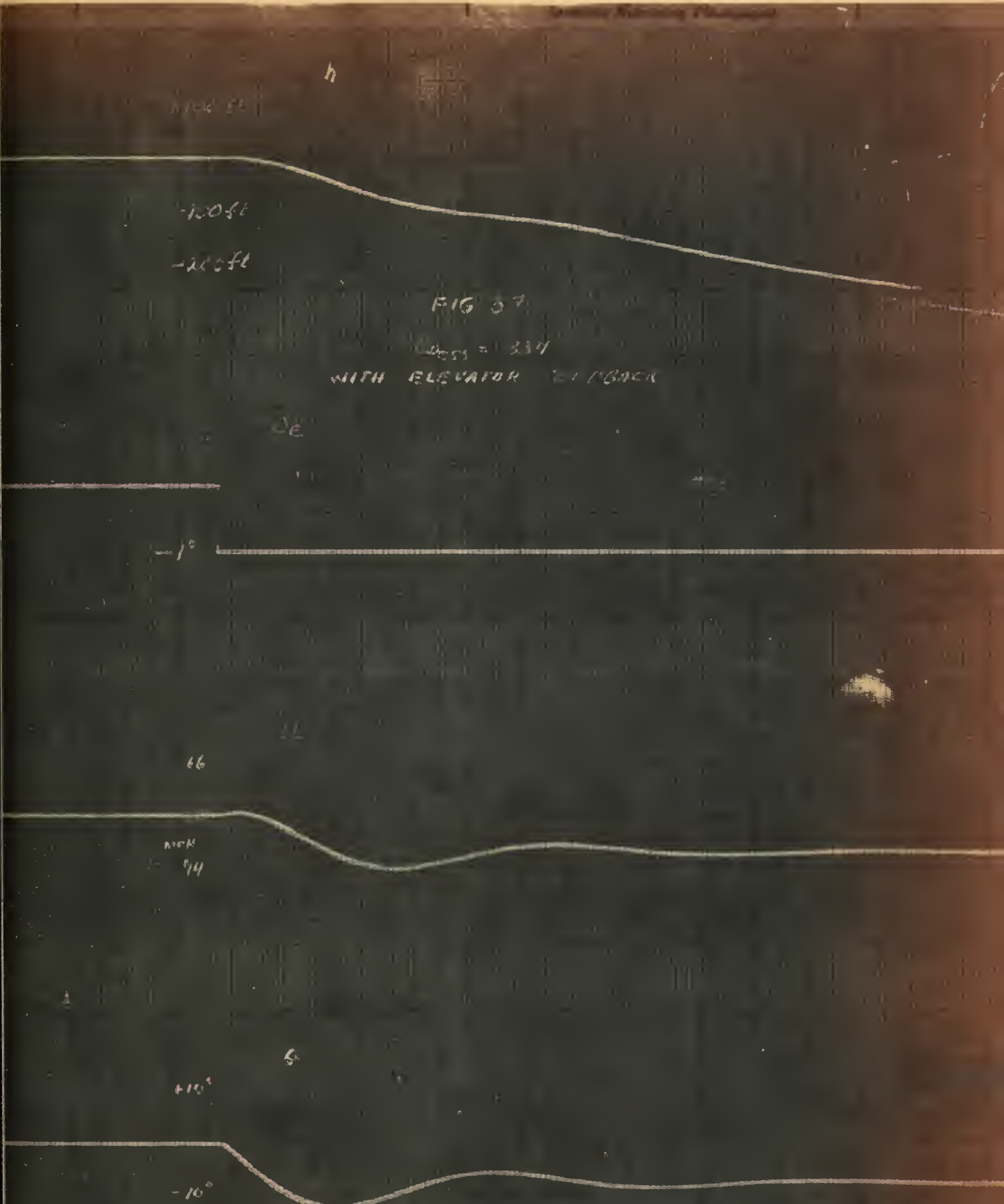
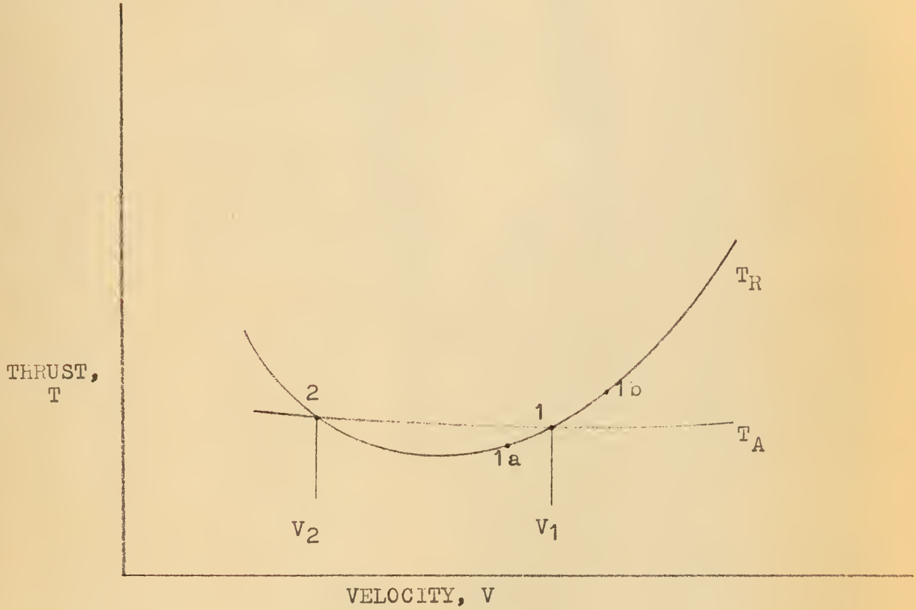


FIG 37

WITH ELEVATOR FEEDBACK

Fig. 38

TYPICAL THRUST CURVE



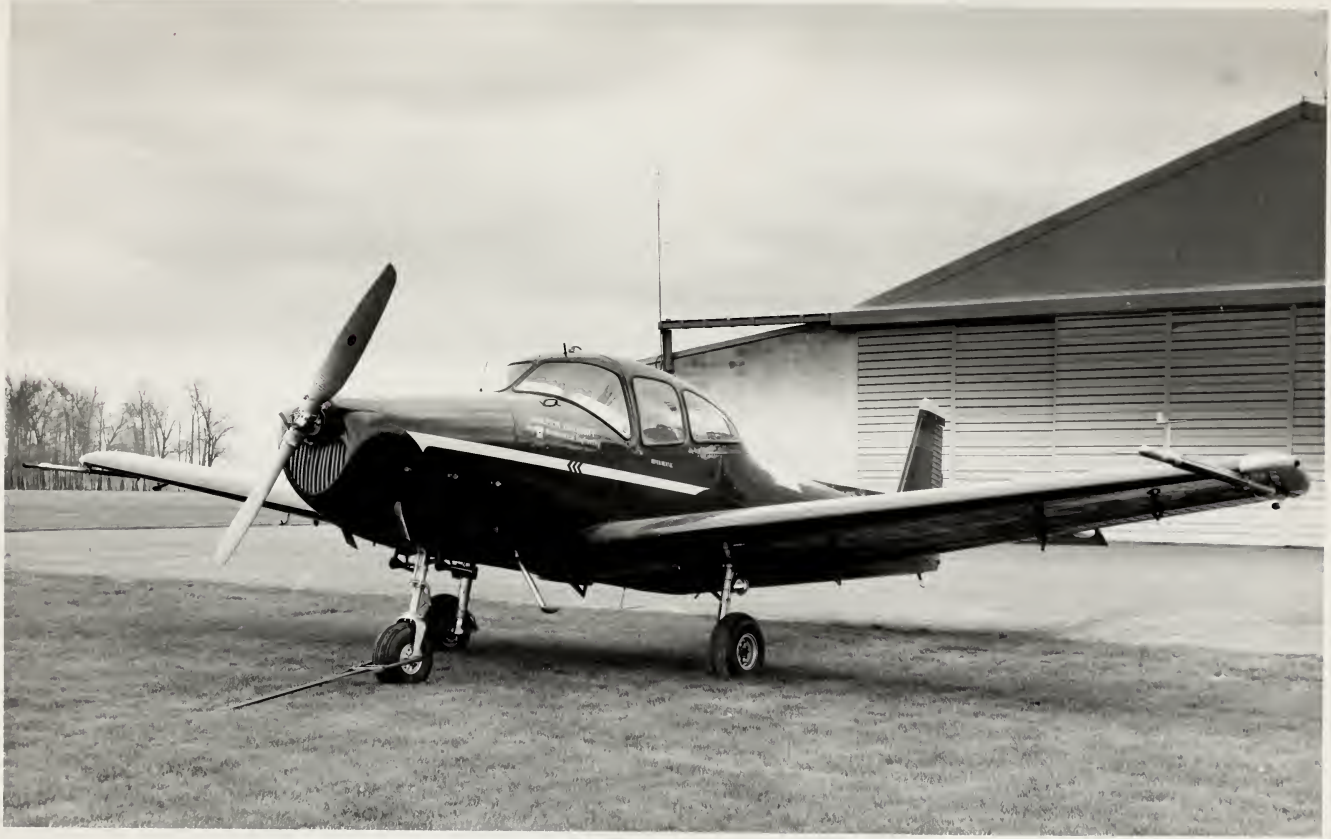


FIGURE 39

EXPERIMENTAL VEHICLE

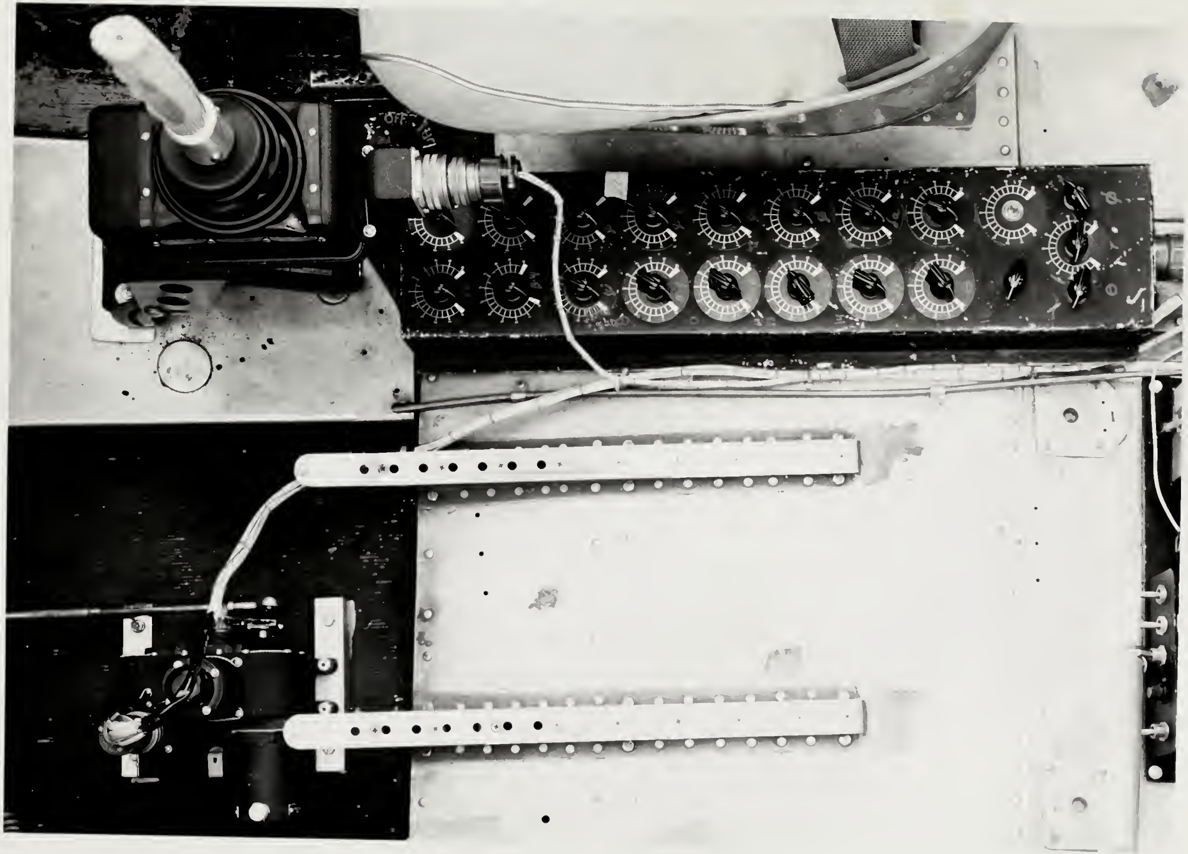


FIGURE 40

COCKPIT INTERIOR SHOWING THROTTLE SERVO,
ELECTRIC THROTTLE AND POTENTIOMETER BOARD

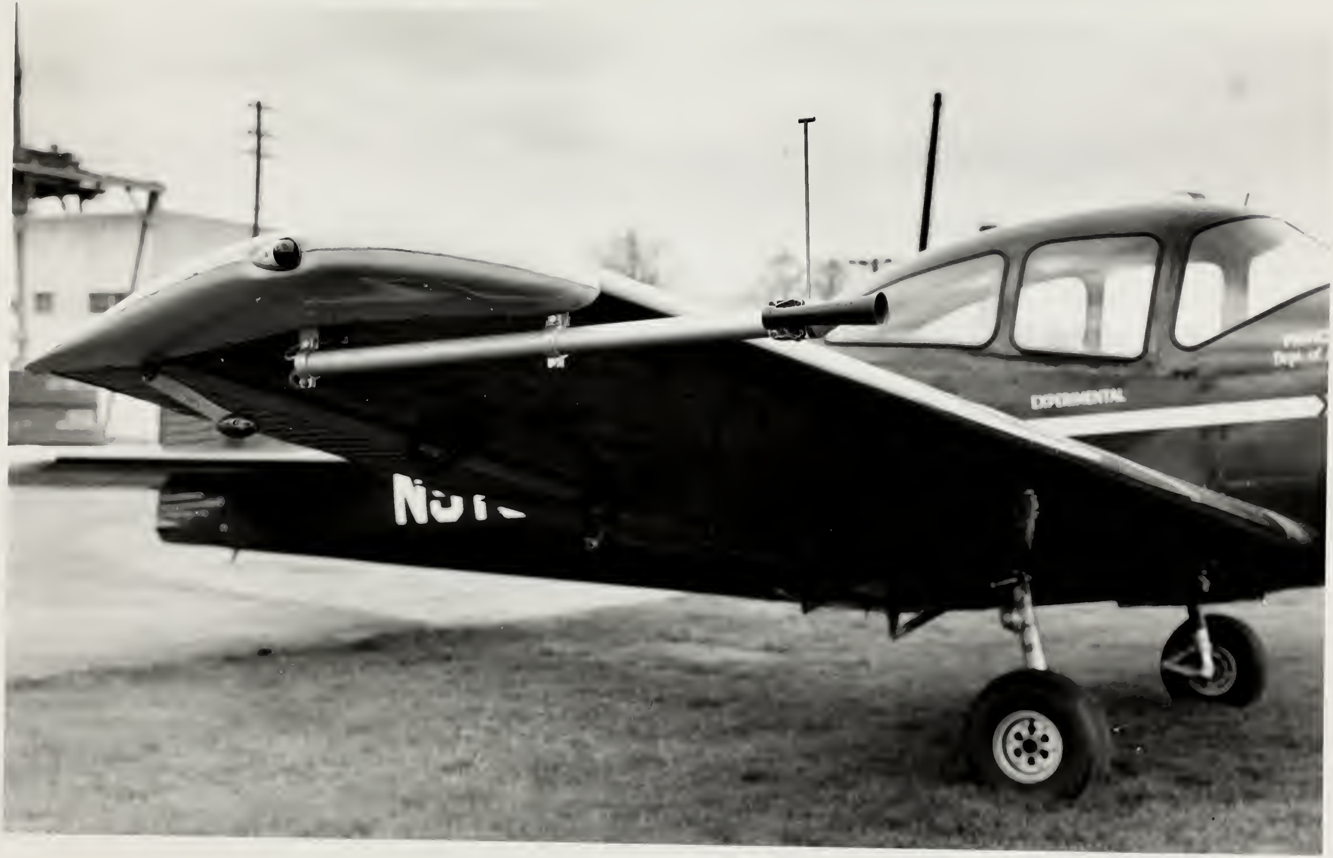


FIGURE 41

STATIC AND PITOT PRESSURE SENSOR

FIG. 42

ALTITUDE RESPONSE TO STEP ELEVATOR DEFLECTION AT DIFFERENT $C_{D_{eff}}$

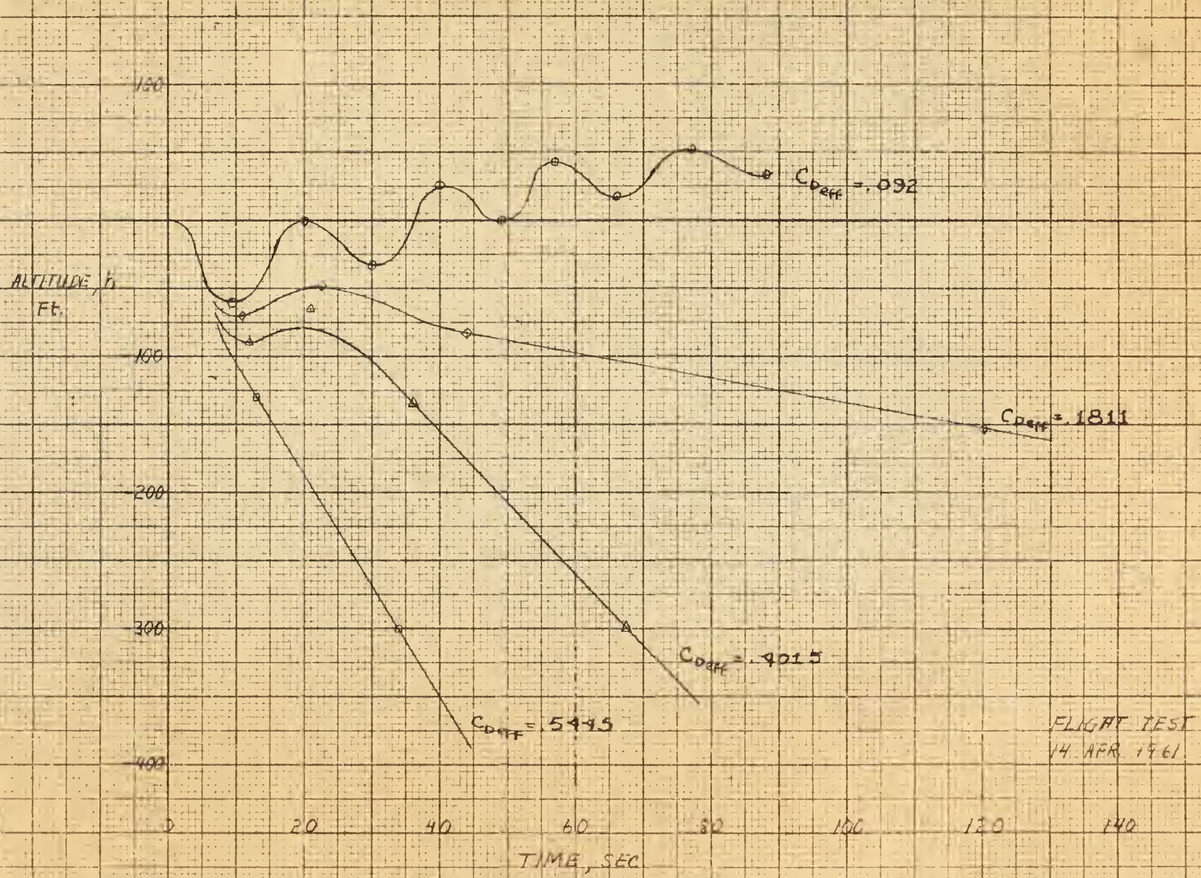
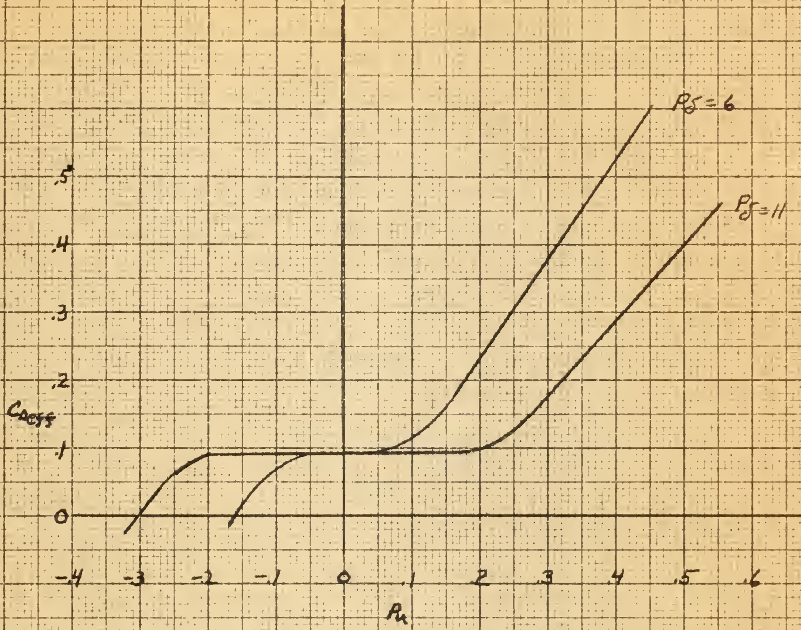


FIG. 43

CALIBRATION CURVES



MANUAL THROTTLE and ELEVATOR

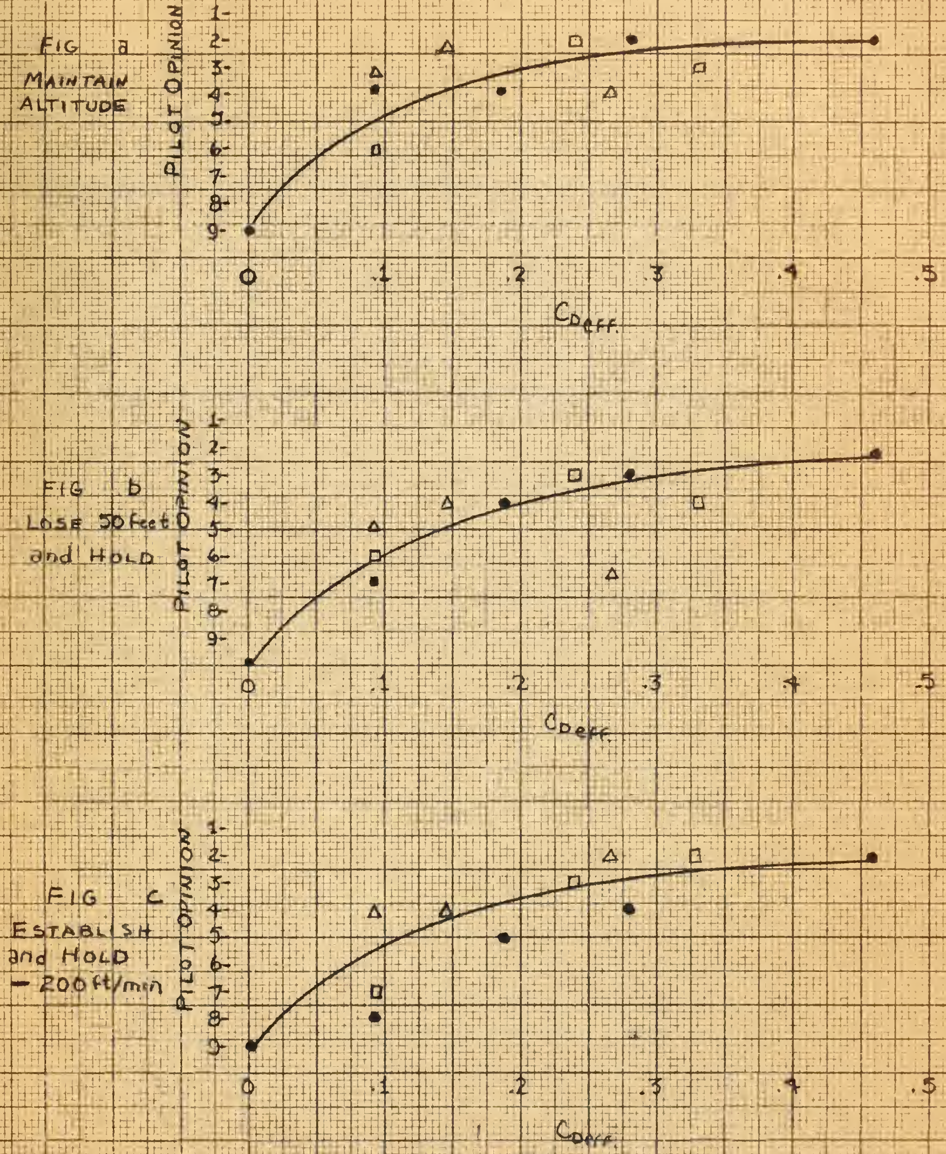


FIG a MAINTAIN ALTITUDE

FIG b LOSE 50 feet and HOLD

FIG c ESTABLISH and HOLD − 200 ft/min

FIG 44

PILOT OPINION versus $C_{D_{eff.}}$

ELEVATOR ONLY

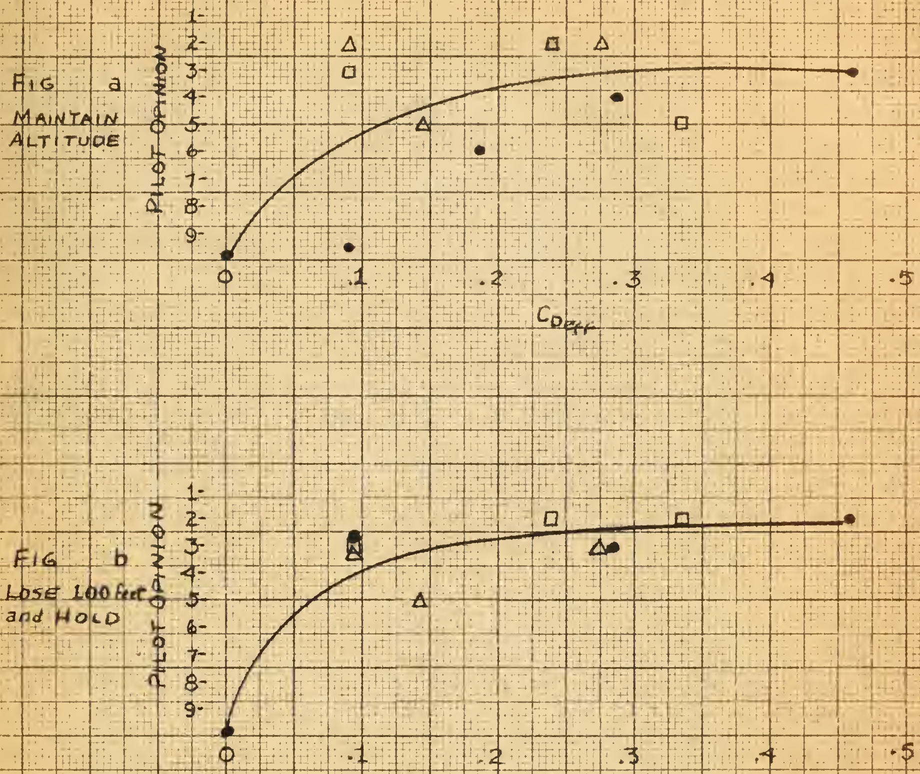


FIG a MAINTAIN ALTITUDE

FIG b Lose 100 feet and HOLD

FIG 45

PILOT OPINION versus $C_{D_{eff}}$

FIG. 46

COMPUTER MECHANIZATION FOR SOLUTION OF ALTITUDE

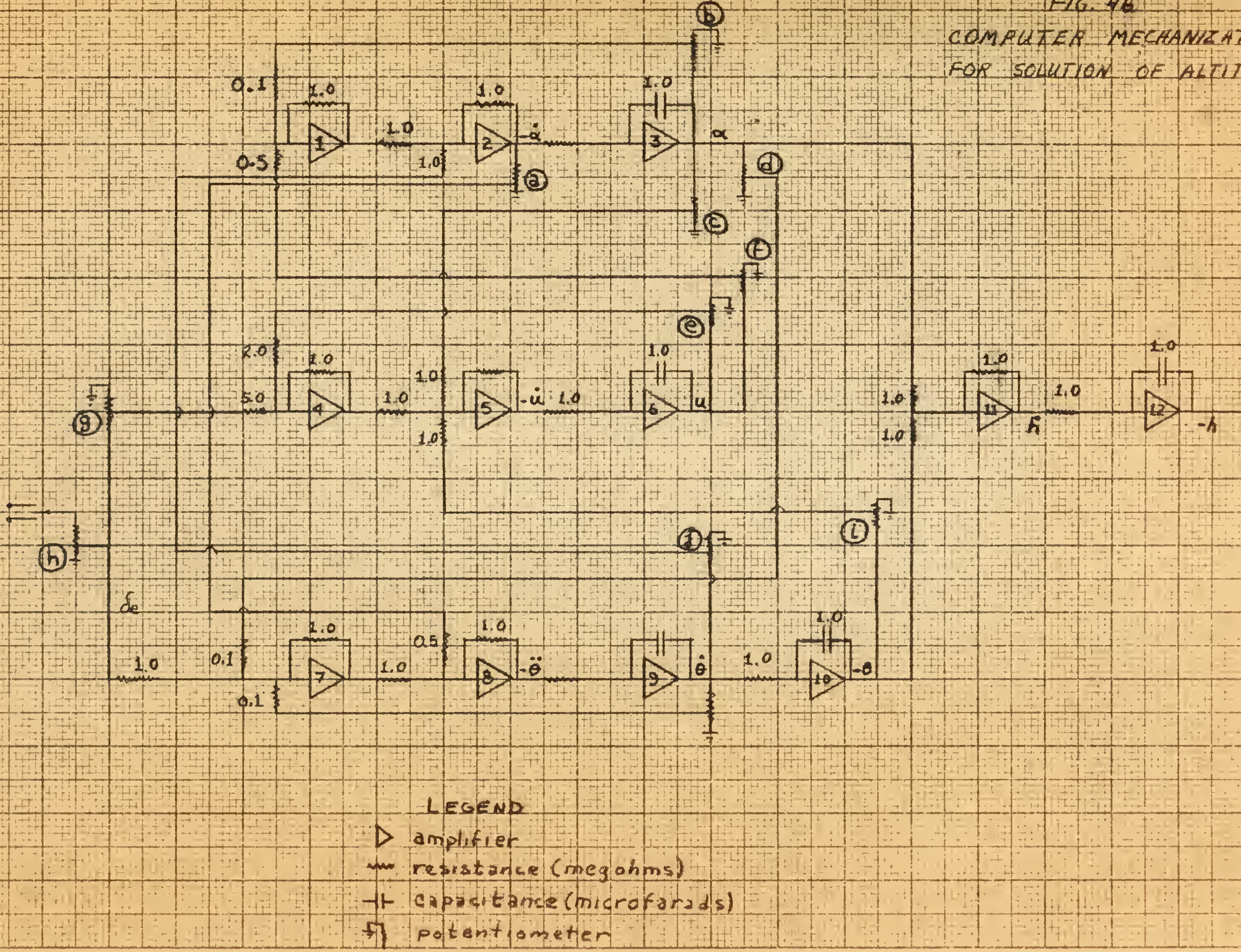


FIG. 47

CONTINENTAL SERIES "E" ENGINE

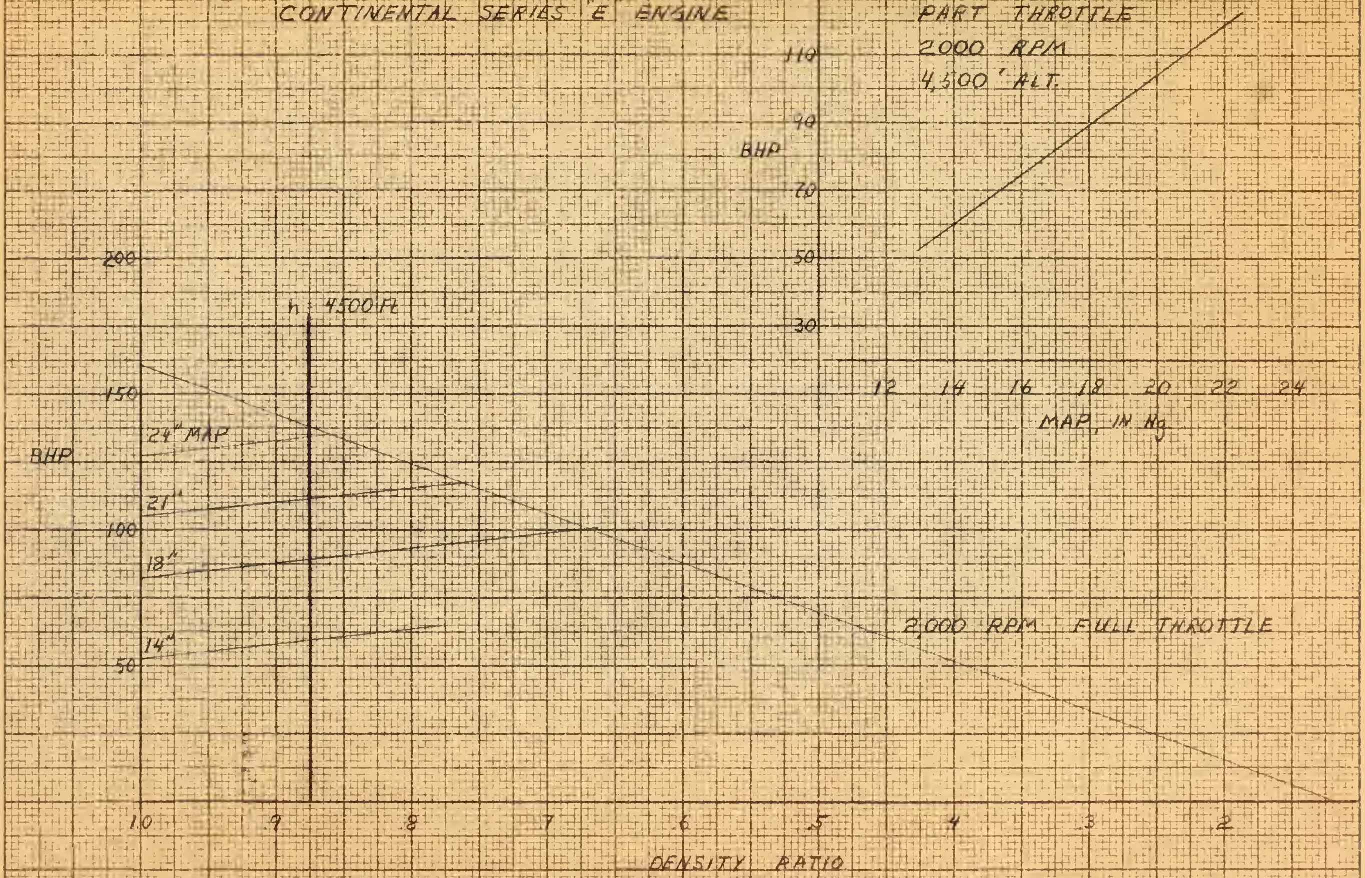
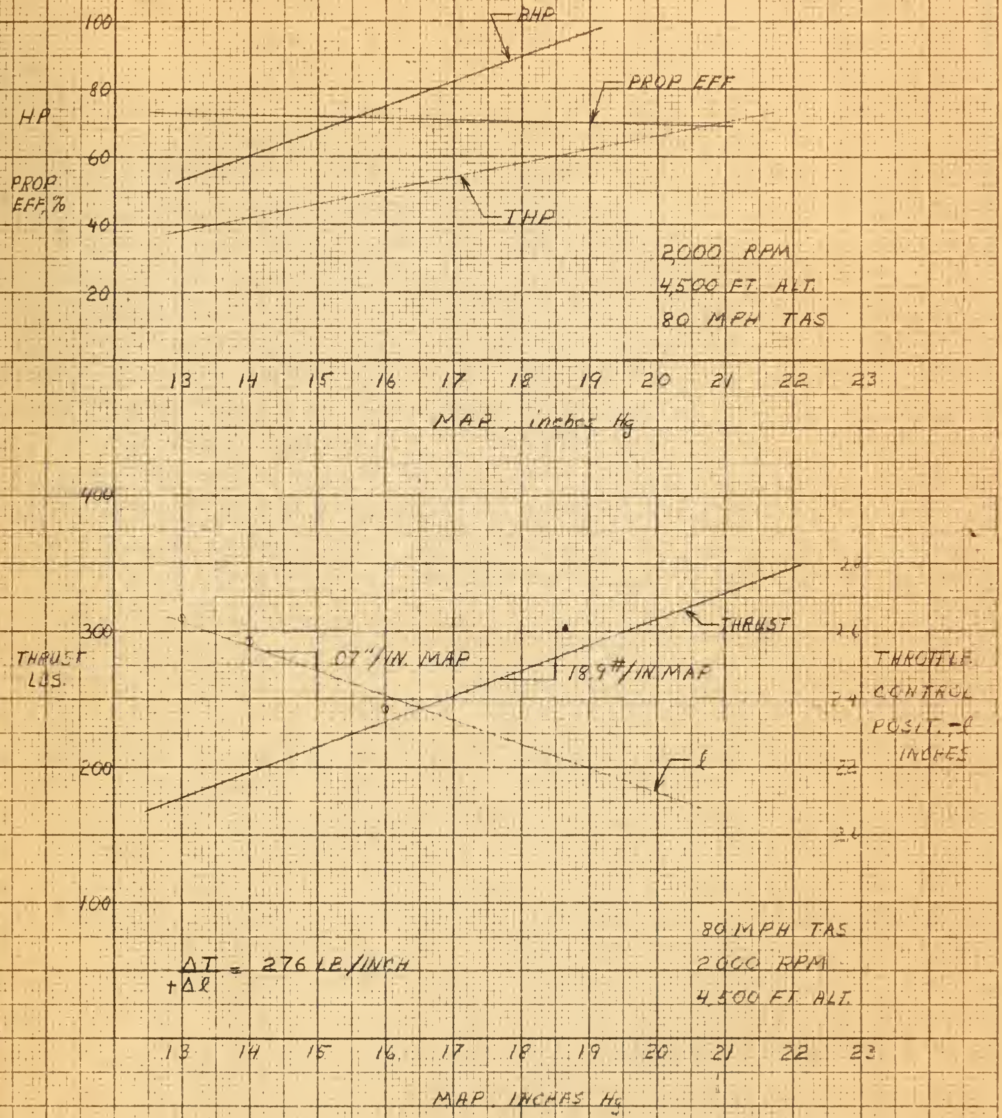


FIG. 48

CONTINENTAL SERIES "E" ENGINE



thesS52
An investigation of the influence of alt
3 2768 001 91437 7
DUDLEY KNOX LIBRARY